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NASA PROJECT APOLLO WORKING PAPER NO.

PROJECT APOLLO

TECHNICAL LIAISON GROUP MEETINGS - SELECTED EXCERPTS [U]

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AVAILABLE TO NASA HEADQUARTERS ONLY

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

SPACE TASK GROUP

Langley Field, Va.

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TABLE OF CONTENTS

LIST OF FIGURES

INTRODUCTION

A series of Technical Liaison Groups have been formed for intra-NASA exchange of technical data related to the Apollo project. This paper presents a series of selected excerpts from the minutes of the January 1961 meetings of these groups believed to contain technical data of particular interest. It should be noted that in many cases the discussions presented are not complete, that some of the concepts presented evolved during the course of the meeting and are subject to further checking, and the numerical data included may be in error. Extreme caution should be used in applying such data. On the other hand, it is felt valuable to make this information available at the earliest possible date, in order to expedite progress in the Apollo studies by making those persons concerned with the studies aware of the general value, extent and availability of the work being undertaken within the NASA.

The membership is composed entirely of NASA personnel from Space Task Group, the various research and flight centers, the Jet Propulsion Laboratory and NASA Headquarters.

The excerpts are presented in a separate section of this report for each Technical Liaison Group. All referenced figures, tables, and report titles will be found in or at the end of that section in which they are referenced rather than at the end of this paper. The numbering systems will be found to be consistent within each section, but not necessarily between sections.

There are nine Technical Liaison Groups:

1. Trajectory Analysis
2. Instrumentation and Communication
3. Mechanical Systems
4. Heating
5. Guidance and Control
6. Configurations and Aerodynamics
7. Human Factors
8. Onboard Propulsion
9. Structures and Materials

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Note: ARC - Ames Research Center
FRC - Flight Research Center
JPL - Jet Propulsion Laboratory
STG - Space Task Group
LRC - Langley Research Center
LeRC - Lewis Research Center
MSFC - Marshall Space Flight Center
OLSP - Office of Life Science Programs, NASA Headquarters

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SECTION I

EXCERPTS FROM

TRAJECTORY ANALYSIS

APOLLO TECHNICAL LIAISON GROUP MEETING

January 21, 1961

Ames Research Center

Moffett Field, California

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Space Task Group (STG).-

a. Summary report.-

(1) Longitudinal range study for reentry at near-escape speeds.- A study is in progress to show the general characteristics of the longitudinal-range capabilities for reentry at near-escape speeds. The controllable longitudinal range is defined and the longitudinal range and controllable range determined for a range of L/D's from 0.2 to 0.7 and W/C_DA's from 25 to 100. The study assumes reentry from equatorial west-to-east orbits and constant aerodynamic characteristics for the reentry vehicle with both Mach number and angle of attack. The results to date indicate that considerable longitudinal range control can be obtained with vehicles having low L/D's at the expense of reducing the corridor depth and bouncing out of the atmosphere. For an L/D of 0.35 and W/C_DA of 50, the range to touchdown from 400,000-foot altitude could be controlled from a maximum of 5,760 miles to a minimum of 1,440 miles with a corridor depth of 30 miles.

(2) Lateral range studies for reentry at near-escape speeds.- A study is in progress to show the lateral range capabilities for reentry at near-escape speeds for reentry vehicles having L/D's of 0.35 and 0.5 and W/C_DA of 25, 50, and 75. The study is for equatorial reentry west-to-east and assumes aerodynamic characteristics constant with Mach number and angle of attack. Results to date show that for constant bank angles initiated before reentry, the lateral range is a function of both longitudinal range and corridor depth. For an L/D of 0.35, a maximum lateral range of 350 miles was obtained at a longitudinal range of 4,350 miles for a corridor depth of 30 miles. It was noted that at longitudinal ranges of 12,000 miles there was little or no lateral range.

(3) Manned lunar landings from lunar orbits.- A study is in progress to determine the velocity and fuel requirements necessary to make a soft lunar landing from lunar orbits of various altitudes. The study will consider both horizontal and vertical contact. Preliminary results indicate that the incremental velocity changes required to descend from low-altitude orbits (50 to 100 nautical miles) is not greater than that for soft lunar landings from collision trajectories. However, descent from orbits with apocynthions of 1,000 nautical miles requires up to 1,000 ft/sec more retrovelocity.

b. Reentry control study.-

One of the requirements of a satisfactory space vehicle is that it be able to return safely to earth to a predetermined point. In order to accomplish this point return, it is necessary that the space vehicle be designed with enough range control on reentry to correct for variation

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such as navigation errors in the return position and for errors during reentry resulting from variation in the earth's atmosphere. To obtain information on the range control capabilities of lifting vehicles, a study was made of the effects of L/D and corridor width on the controllable range. Controllable range is defined as the overlap in the ranges that can be obtained at the top and the bottom of the corridor, as illustrated in figure 1. The bottom of the corridor is defined by the steepest reentry which will not exceed a given maximum deceleration. The trajectory at the bottom of the corridor is obtained by holding maximum positive L/D until the maximum deceleration point is passed after which the lift can be modulated to provide range control. For the shortest range, the maximum deceleration is held constant until nearly touchdown. For maximum range the L/D was held constant until touchdown.

The top of the corridor is commonly defined by the trajectory which reenters in one revolution around the earth when full-negative L/D is applied; that is, the boundary between multiple- and single-pass returns. This definition in itself states that the minimum range at the top of the corridor is 25,000 miles. It is apparent, therefore, that with this definition for the top of the corridor, controllable range can only be obtained at high L/D's.

The first step in the range control study, therefore, was to examine the range characteristics near the top of the corridor. Figure 2 shows the results of the analysis of the range for full-negative L/D held constant to touchdown. It is observed that as the reentry angle is decreased, the range increases slowly and then brakes very sharply. It is observed that the range jumps from about 20° or 1,200 miles to 360° or 25,000 miles with less than 0.1° change in reentry angle. The top of the reentry corridor can be defined to correspond with this brake in the range curve with the loss of less than 1 mile or 0.1° in corridor depth. We gain, however, a considerable overlap in the ranges at the top and bottom of the corridor.

With this definition of the corridor in mind, an analysis was made of the controllable range; that is, the difference between the maximum range at the bottom of the corridor and the minimum range at the top for range L/D's and maximum deceleration levels. This study was made for equatorial reentries in an easterly direction. No restrictions were placed on the vehicle in respect to the altitude of bounce as it was desired to analyze the maximum capability.

The results of the study are shown in figure 3. There is presented in figure 3 the controllable range as a function of corridor depth for L/D from 0.3 to 0.7 and maximum total deceleration from 4g to 10g. The results show that for L/D of 0.4 and a maximum deceleration of 6g, the controllable longitudinal range would be about 13,000 nautical miles

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and the corridor about 28 miles. It is apparent that considerable controllable range is available in L/D region of 0.3 to 0.4 even at the high g reentries.

The results shown in figure 3 are for $\frac{W}{C_D A}$ of 100. Preliminary results for $\frac{W}{C_D A}$ of 50 indicate that the general picture is about the same. However, there is likely to be some increase in the controllable range at the lower ballistic number. The results at a ballistic number of 50 are now nearing completion.

In general, the long ranges are associated with a skip off the top of the atmosphere. This skip at times results in a bounce at very high altitudes. Since there is some concern as to additional passes through or into the radiation belt, an indication of the altitude of bounce is shown in figure 3. The vertical line represents the conditions for a bounce to 500-nautical mile altitude. Ranges to the right of this line are associated with a bounce to higher altitude with the extreme range at $L/D = 0.7$ going to 4,500-nautical mile altitude. Ranges to the left of the vertical line are associated with bounce altitude of less than 500 nautical miles. For 10,000 controllable range, the bounce reaches an altitude of less than 250 nautical miles.

Ames Research Center (ARC).-

a. Summary report.-

(1) Trajectories.- The abort trajectory work was summarized using a set of figures which illustrated V requirements as a function of various parameters for returning to the earth as any time during boost. This work was done by Robert Slye of the ARC 3.5-Foot Tunnel Branch.

The atmospheric-entry work reported was on the "Effect of Lateral and Longitudinal-Range Capability on Reentry 'Window' for Lunar Mission." This work illustrated means of defining tradeoffs between midcourse-guidance accuracy and L/D. The work was done by Al Boissevain of the ARC Supersonic Free-Flight Tunnel Group.

The circumlunar trajectory work reported referred to a paper entitled, "Some Characteristics of Ballistic Circumlunar Trajectories." The work was done by the Guidance and Control Branch.

Some results regarding implications of direct ascent and point return on characteristics of trajectories were reported. This work was done by members of the Guidance and Control Branch.

(2) Navigation.- The navigation work reported was in the two areas of midcourse and entry. The discussion on midcourse area was as follows:

(a) Errors in determining position along a reference trajectory for three optical instrumentation methods.

(b) Circumlunar navigation accuracy using Kepler's Laws of Orbital Motion.

(c) Optimal smoothing for optical measurements.

(d) Velocity and position sensitivities for certain reference trajectories. No written reports are available on subjects (3) and (4) at the present time. Rough block diagrams and graphs were used to describe the subjects. The above work represents some current efforts of the ARC Guidance and Control Branch.

(e) The atmospheric entry navigation work reported was on a system which is under study by the Flight and Systems Simulation Branch. The work was done by R. Wingrove of ARC Flight and Systems Simulation Branch.

b. Effect of lateral and longitudinal range capability on reentry "Window" for lunar mission.-

Figures 4 and 5 are presented to illustrate, for a target latitude of 35°, the allowable variations in the time of reentry and/or orientation of the orbital plane for various latitudes of vacuum perigee.

The following is a brief explanation of the geometry, methods, and assumptions of a preliminary analysis of one of the problems of point return from the lunar mission.

The geometry and notation used are shown in figure 4. This figure represents the northern half of the celestial sphere on which is superimposed the orbital plane. The vacuum perigee and the point of entry are shown on the orbital plane. β is the angle between the equatorial plane and the orbital plane. At values of β less than the target latitude, there are no points in common between the orbital plane and the target latitude.

The target latitude represents the locus of positions of the target during the course of a day. The position of the target can be defined by the latitude of the target and the elapsed time since the target was in the plane of the zero-right ascension. (The celestial sphere does not rotate with the earth. A line from the center of the earth toward the star Aries and the polar axis defines the plane of zero-right ascension.)

The vacuum perigee and the orbital plane of the lunar return trajectory are fixed relative to the celestial sphere and are independent of time. The earth rotates, however, making the target's position relative to the vacuum perigee and the orbital plane time-dependent.

It is assumed that the range capabilities of the vehicle can be represented as a long, narrow rectangle of ±500-mile lateral range and a longitudinal range starting at 2,000 miles from entry and extending to 5,000, 7,500, and 10,000 miles from entry. This is a conservative statement of the capabilities of an $L/D = \pm 0.5$ vehicle if the entry velocity is 36,000 ft/sec and if the vehicle is allowed to skip to an altitude of not more than 400 miles. Further work is being done to obtain an accurate boundary of range capabilities for a number of values of L/D limits and values of entry velocity. One further simplification was made; namely, that the earth does not rotate during the period of the reentry. The effect of this simplification is to freeze the relative positions of the target and the vacuum perigee. Removing this simplification will warp the results presented here, but will not seriously affect the conclusions. Entry time for a 5,000-mile range is of the order of 15 minutes.

Still referring to figure 4, the intersection of the locus of possible landing sites and the target latitude describes the possible target positions at which a landing can be made. This variation of possible target sites gives freedom in two dimensions: one is the orientation of the orbital plane; and the other is an allowable error in the time of return, since the location of the target along the target latitude is time-dependent.

The range of allowable orbital-plane-orientation angles and/or the difference in the longitude of vacuum perigee and target can be computed for various values of vacuum perigee latitudes. In figure 4 the difference in these longitudes is defined at $T_t - T_p$. This difference is positive when the target longitude is displaced, as shown relative to the perigee latitude, when the target is east of the vacuum perigee.

Since the earth rotates at a rate of 15° per hour, the angular difference in longitude is expressed as an equivalent difference in time in the rest of the analysis.

For the assumed target of 35° North Latitude, figure 5 shows the allowable values of the orbital plane orientation and effective time difference between the target and vacuum perigee that will permit the vehicle to intersect the target. The centerline of the plot represents the locus of these points for a vehicle with no lateral range capability. The area between the curves shows the allowable values assuming a vehicle with a lateral range of ±500 miles. The hatched boundaries are the

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restrictions imposed by the available longitudinal range. Entry was assumed to occur on the orbital plane 10° before vacuum perigee is reached. This corresponds to an entry angle of 5°.

One example is described. Assume that the vacuum perigee will be at 20° North latitude and at a given longitude in the celestial sphere. (The actual value of the longitude has no bearing on the solution, since a relative difference in longitude is determined.) Assume further that the orbital plane is oriented at 40° from the equatorial plane. Question: What will be the allowable error in time of entry for a target at 35° North latitude? Answer: From figure 5(c), at a β of 40° and looking first at the condition of no lateral range capability, the centerline of the plot. The target longitude must be 2 hours in advance of the vacuum perigee longitude. Because of the earth's rotational rate of 15° per hour, this is equivalent to a displacement of the target longitude 30° to the east of the vacuum perigee longitude. The target can also be 6.7 hours in advance, or 100.5° further to the east. Note that the curves are cyclical with a period of 24 hours. With the lateral and 5,000-mile longitudinal range capability described previously, the allowable target longitude can be anywhere from 0.8 hour in advance of the vacuum perigee to 5 hours in advance - that is, from 12° to 75° further eastward. If the longitudinal range is increased to 7,500 miles from entry, the target position could lead the perigee by 8 hours, or 120° further to the east. At the same time, the allowable orbital plane orientation, β, can vary between limits. If, at entry, the target is 2 hours in advance of the vacuum perigee, then β can have any value from 30° to 54°. When the target longitude is west of the vacuum perigee longitude at the time of entry, the orbital plane orientation must be greater than 90°, as is indicated.

c. Abort-rocket requirements for escape trajectories.-

Failure of the propulsion system for the manned lunar mission at velocities greater than satellite speed may leave the lunar vehicle on a highly elliptical orbit with subsequent long exposure time of the crew to the radiation belts. Direct retrothrust at escape velocity is clearly expensive in terms of rocket thrust as over 10,000 fps is required to reduce the velocity to satellite speed. Consider instead a direct return to earth by using the abort rockets in such a manner as to deflect the trajectory into the earth's atmosphere. If the boost trajectory is low, the abort-rocket requirements are considerably less than those required for direct retrothrust, and reentry into the atmosphere is similar to a normal return from the moon at escape speed along the overshoot boundary.

Figure 6 shows a typical three-stage boost trajectory which reaches escape velocity at 500,000-foot altitude with a horizontal flight path. Note that the altitude reached earlier in the trajectory is considerably

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higher than the burnout altitude. This is characteristic of booster systems which have a long burning time and results in a burnout point at a low altitude from which a direct return can readily be made. An abort rocket ΔV of the order of 4,000 fps applied in the direction indicated on the figure is sufficient for reentry along the overshoot boundary for a reentry vehicle of $m/C_DA = 5$ and $L/D = 0$ and the reentry is completed in a single pass.

Figures 7 to 10 show the effect of the reentry vehicle parameters m/C_DA and L/D as well as the effect of the rocket flight conditions at burnout. The ΔV required is reduced if the vehicle has a high-drag loading or uses negative lift, but the effect is small in comparison with the importance of keeping the altitude and the flight-path angle low at burnout. For nominal reentry vehicle and a boost trajectory with burnout at escape velocity with a zero flight-path angle, the abort rocket ΔV required increases rapidly with burnout altitude and is approximately 2,000 ft/sec at 300,000 feet, 4,000 ft/sec at 500,000 feet, 6,000 ft/sec at 900,000 feet, and 8,000 ft/sec at a burnout altitude of 1,500,000 feet. The penalty for a positive flight-path angle at burnout is over 500 ft/sec per degree.

There is a lower limit to the altitude at burnout since drag effects at escape speed may be significant if the boost trajectory traverses lower altitudes. This has been found to be about 300,000 feet.

Figure 6 also shows an abort reentry at suborbital speeds; the so-called maximum g abort. The critical area in the boost trajectory tends to occur in the velocity range from 14,000 to 16,000 ft/sec where the subsequent reentry decelerations are severe if uncorrected. The optimum point for firing the abort rocket is not at the fail point but just before reentry. For the example shown, a ΔV of 3,000 fps is sufficient to reduce the reentry deceleration from 18g to 8g for a reentry vehicle of $L/D = 1/2$. However, it has been found that the maximum g abort-rocket requirements are usually within the capability of the required ΔV for the escape-velocity point.

d. Atmospheric-entry navigation simulation.-

This report concerns systems to guide maneuverable vehicles to a desired touchdown point through a planetary atmosphere without exceeding arbitrary temperature and acceleration limits. The concept of using perturbations about a fixed or stored trajectory has been considered in many studies, figures 11 through 14, for example. These studies have shown that successful touchdown control with good precision can be achieved if the actual initial conditions of the entry are sufficiently near the stored values and if enough perturbation variables are used. The fixed-trajectory method is inherently limited, however, to the conditions and situations stored in the system.

Guidance systems for advanced missions may have to cope with quite wide variations in abort and entry conditions as well as nonstandard and different atmospheres. This has stimulated interest in the possibilities of more "universal" concepts that do not depend on stored conditions but rather continuously compute or predict the future trajectory from the present actual conditions. In addition to greater generality, a continuous prediction concept might possess advantages in deriving and displaying hazardous flight regimes and in computing the total maneuvering capability to enable the controller to decide upon alternate flight paths or destinations more easily.

From experience with fire-control systems, it is inferred that one of the major problems in a continuous prediction system will be the form of the equations used as a basis for the prediction computer. A satisfactory compromise must achieved between the conflicting requirements for a reasonable amount of computing equipment, accuracy, speed, and realistic input information. To gain insight into these questions, a research program has been conducted with three goals: (1) to develop a continuous trajectory prediction technique; (2) to develop a display and control system to use the information generated; and (3) by means of an analog simulation to see how a pilot or automatic control might be used to close the control loop.

For clarity, a brief description of the final system concept is presented prior to the general discussion of the prediction equations, the display and control system, and the simulation results.

Figure 11 is a block diagram of a guidance and control loop using the concepts studied in this report. An inertial platform and navigation computer continuously measures and computes the present flight conditions and destination information to feed to the prediction computer. The prediction computer continuously computes the maximum maneuver capability with respect to the destination (or destinations), and feeds this information to an automatic control system and a pilot's display. Either the automatic control system or the pilot, through an override, controls the vehicle to acquire and keep the destination in the center of the maximum maneuver-capability envelope.

The elements considered in this report are the prediction computer and display. The function of the prediction computer is illustrated by figure 12. A typical reentry phase is used for convenience. From the present vehicle condition, trajectories are computed for three constant trim values: maximum, minimum downrange, and maximum crossrange which are used to define a "footprint" showing maximum maneuver capability. The important point is that the differential equations of motion for the trajectories are solved by a "fast" computation in the airborne computer so that repetitive solutions are made continuously for the changing flight conditions.

The desired destination is also referred to in figure 12. This information is located with respect to a nondimensionalized maximum maneuver boundary and presented to the pilot by the display shown in figure 13. Thus, the pilot, by noting the destination with respect to his maneuver capability, can make a choice of control inputs.

Limits may be imposed on the maneuver capability of a particular vehicle in the form of acceleration or heating boundaries or in the form of conditions where the vehicle will "skip" out of the atmosphere in an undesired manner. With the repeated prediction of total trajectories by fast computation if any of these limits are encountered, this can be indicated on the display as in figure 13 or fed to the automatic control system.

A guidance system for maneuvering vehicles within a planetary atmosphere has been developed using the concept of fast, continuous prediction of the maximum maneuver capability from present conditions rather than a fixed-trajectory technique. A method of display and control was developed which compares desired touchdown points with the maximum maneuver capability and heating or acceleration limits, so that a proper decision and choice of control inputs can be made.

A piloted analog simulation was used to demonstrate the feasibility of the concept and study its application to control of lunar-mission reentries and recoveries from aborts. A repetitive solution time of 5 seconds was adequate for reentries from satellite speeds where conditions were not changing rapidly, but for lunar missions a faster solution time will be necessary. The regions of entry conditions leading to control-sensitivity problems were defined. The simulation was also used to define the ground areas that would be attainable during typical entries using this method of guidance control for a vehicle with moderate lifting capability ($L/D = 0.5$ - fig. 14).

Marshall Space Flight Center (MSFC).- Due to the strong interrelationship of trajectory analysis with such areas as vehicle control, path guidance, structural and heating constraints, pilot-abort requirements, launch restrictions, and mission requirements, the approach at MSFC to the trajectory problem is a procedure of continuously increasing integration of the requirements into the flight profile, to arrive finally at optimum solutions.

The following list of fields of activities points out this endeavor. A full integration, however, is not possible, since some essential factors have not been defined (e.g., reentry body).

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a. Flight analysis.-

For the Saturn C-1, as well as the C-2, continuous trajectory analysis is going on, incorporating the latest information. All studies are made by means of calculus-of-variation methods. Profiles are established for:

(1) Circular-orbit missions (C-1 and C-2)

(2) Lunar missions (C-1 and C-2)

(3) Reentry test missions (C-1)

(4) Block I and Block II missions (C-1)

The following variations are studied where applicable:

(1) Variations of injection altitude

(2) Variations of minimum altitude during high-velocity history (for heating)

(3) Variations of injection-path angle (for lunar missions in view of abort studies)

Three-stage flights, as well as four-stage flights, are being studied. For all missions, the best ratio of propellant loading is being investigated. Special consideration is given: (1) to the vehicle-control limitation during first-stage flight, (2) to the separation problem at the end of first-stage flights, and (3) to the engine-out situation in the first-stage flight. Also, viewpoints of flight interruption by circular-parking orbits are investigated, for achieving simultaneous optimization of azimuth and pitch of injection-velocity vector. Studies are made toward optimum operational schemes and associated propellant requirement for aborts from the powered phases leading toward circular orbits (C-1) and lunar flights (C-2). Trajectories with various end conditions as to path angle and altitude are investigated for the lunar mission. Data related to these studies are available and a related publication is in preparation. For the early reentry test flights (SA-7, SA-8, SA-9, and SA-10) the analysis covers investigations of best flights to different reentry point ranges. The loss of payload with increasing range has been established. (A report is in preparation.)

Circumlunar flights have been established on two celestial models:

(1) On the Jacobian restricted three-body model, the characteristics of coplanar flights have been studied, especially to establish

the sensitivity of return conditions to injection variations. This model also has been used to study the efficiency of corrections during any time in flight for corrections of return conditions. Also, the effect of uncertainties of the astronomical constants on the return accuracy was investigated. An estimate of total propellant requirements for midcourse corrections to deal with injection errors and the astronomical uncertainties was reached. (Report No. MNN-M-AERO-3-60).

(2) On the true ephemeris model for a particular launch time, (early 1964), a specific three-dimensional circumlunar trajectory has been established, taking into account the geography of launchings from the Atlantic Missile Range and of return between the Atlantic Missile Range and Ascension Island. Influence factors for injection errors and a possible correction scheme has been discussed. (Report is in preparation.)

An investigation has been started of possible schemes and associated propellant requirements for immediate return at any point on the circumlunar flight.

In connection with the planned, unmanned landings on the lunar surface, as being prepared by Jet Propulsion Laboratory (JPL) powered ascending trajectories and also free-flight trajectories have been established for a particular time of the mission. Coefficients of sensitivity have been established. Landing schemes for unmanned flights have been proposed. (Report will be submitted to NASA Headquarters in the spring of 1961, combining MSFC and JPL activities.)

A comprehensive analysis of atmospheric reentry flights for initial velocities appropriate for return from circumlunar flights had been followed through. Body characteristics were covered from W/C_DA = 50 lb/sq ft to 500 lb/sq ft, with maximum L/D of 0.5. Constant lift, as well as lift modulation, was applied. For maximum deceleration limits and skip limitation, the tolerance of reentry angles was established in reference to a reentry-point altitude. Range variations achievable by maximum lift variations in the limits of specified deceleration and skipping altitudes were established for each reentry angle. Also, the exchange of cross-range corrections against longitudinal corrections was numerically studied. The area of landing points, common to all reentry trajectories of varied reentry angles, reflects back to an allowable spread of reentry points in longitude and latitude associated to the full reentry angle regime (3°). Widening out the geographical area of the reentry point requires a narrowing of allowable reentry angles. The consideration of their mutual relationships leads to the definition of the "reentry manifold," a surface in the parameters of position vector, velocity vector and time. This manifold is then the reentry criterion (or return window) that governs the midcourse guidance mode, as discussed in the following chapter under guidance. (Report No. MNN-M-AERO-4-60).

The layout of the powered phase of the reentry test flights has been arranged to end in injection conditions that are identical to the reentry conditions of lunar flights. Landing points can be provided in a regime that contains Ascension Island. (Report is in preparation.)

b. Guidance mode and system studies.-

In close connection with the trajectory analysis, a number of guidance studies are carried on concerning all flight phases of the circumlunar flight. Since all hardware development questions are not the primary concern of this Liaison Group, we list in the following only those studies that aim toward the establishment of a guidance theory or a "guidance mode." Two approaches are going parallel in some flight phases. The first approach is the endeavor to work out a comprehensive guidance theory for best guidance performance, i.e., a guidance mode with an optimization requirement embodied in it, where the optimization function can be specified, as e.g., propellant consumption. The second approach is laying emphasis on simplicity. Systems of the second type may be thought of as being used as secondary or emergency systems or they may be programed in early flights in case the more comprehensive system is not yet available.

The length of the powered flights of the Saturn vehicle (in time or in range) and the magnitude of expected variations due to engine-out situations in first-stage have as yet frustrated all efforts to develop guidance modes that are (a) simple in the sense that flight constraints can formulate before flight to which the vehicle is to be path controlled, and (b) not using an excessive amount of reserve propellants. Therefore, studies here are proceeding toward the development of a novel guidance mode, the so-called "Adaptive Guidance Mode," that incorporates as an essential feature the capability of the system to choose at any instant of flight the most economical path toward the injection manifold. The mathematical tools used for deriving the guidance equations are identically those needed in trajectory analysis, e.g., the methods of calculus of variations. Here it becomes necessary to merge the fields of trajectory analysis, and guidance theory.

The application of the Adaptive Guidance Mode to the powered phase has some implications as to the guidance mode to be used in following phases. Since the adaptive mode is endowed with the intelligence to choose the point of the injection manifold that results in least propellant consumption, it does not adhere to a so-called nominal path, in propelled, as well as free flight, but one that insures acquisition of final desired conditions, e.g., the return manifold in circumlunar flight. This feature requires, consequently, from the guidance scheme that is governing the midcourse phase, that it recognize correct or incorrect flight conditions in the light of the acquisition of the end manifold. Only if this is done is the optimum propellant utilization realized. Thus, it follows that also the midcourse guidance mode is to be of the type of "adaptive guidance." This follows then in a restricted sense also for the terminal phase, the atmospheric reentry flight. However, here the optimizing function is not concerned with propellant consumption, but other forms of energy management, e.g., maximum level of deceleration.

The development status of the adaptive guidance mode for application to the Apollo flights is shortly given in the following:

(1) For the reentry test flights the injection manifold has been formulated in mathematical terms and is being used in the steering equation of the powered flight.

(2) Steering equations are being developed for the upper stages of the Saturn flight in a form that can be handled by an onboard digital computer.

(3) The equations for computing the families of optimized reentry flights have been established and are being coded on the IBM-7090 computer.

(4) For the circumlunar flights, the relationships between the end manifold (return conditions) and manifolds representing allowable flight conditions at any time from injection to return are being established by means of the general perturbation methods of astronomy.

Means of hardware arrangements to implement the adaptive guidance mode could be inertial guidance components in powered ascent and reentry flight, and inertial components in conjunction with tracking and command guidance in midcourse flight.

Secondary systems and other problems related to the guidance studies are summarized below:

(1) Concerning the powered phase, simple modes have been developed for pilot abort from flight.

(2) The effect of errors of inertial components of the presently envisioned inertial guidance system on the accuracy of injection conditions is being studied.

(3) For midcourse flight, a linear correction scheme has been studied that formulates a requirement for velocity corrections at linear function of position and velocity deviations at fixed time points. Total impulse requirement for corrections of specified injection deviations was derived from it.

(4) The accuracy of position and velocity determination during midcourse, based on tracking (various combinations of range, range rate, and angular measurements) are being studied and evaluated as to the accuracy at which the reentry point can be either predicted or acquired by means of corrections.

(5) For the atmospheric-reentry, a path-control mode in plane of flight is being studied that controls one space-fixed component of inertially measured path and velocity history according to a preset program. Deviations from the program are controlled out by pitch motion of the reentry body. Atmospheric variations of wind and density variations and also weight variations are controlled out to less than [illegible] (Root-Mean-Square). The mode is rather insensitive to inertial alignment errors.

(6) An emergency system for safe reentry is being studied that is based on altitude and altitude rate measurements at high altitudes (above the conditions of blackout from ionization) and utilizes in lower altitudes measurements of axial component of deceleration. The results are very promising.

<u>Langley Research Center (LRC)</u>.- The following notes give the status of studies at LRC which relate to Apollo, as of mid-December 1960.

An analysis has been performed and a report is being written on the effect of dynamical and geometrical constraints on lunar trajectories. The analysis considers direct ascent, coasting orbit and parking-orbit launch procedures. Restriction on launch-time frequency, daily launch-time interval, lunar position and angle between the earth-moon and earth-vehicle planes are calculated for each launch procedure and for the constraints considered. The constraints result from consideration of AMR safety requirement, guidance and navigation, and radiation dosage resulting from the Van Allen belt.

A paper presenting some results of preliminary trajectory studies which are useful in the design of nominal trajectories for the circumlunar mission is nearing completion. Part of the paper describes the results of a parametric study defining characteristics of trajectories which circumnavigate the moon and return to the atmosphere of the earth with reentry conditions suitable for manned missions. Injection and midcourse guidance studies include an error analysis and calculations of the effects of guidance corrections at various points throughout two nominal trajectories. Some considerations are given to the effect of the return point at the surface of the earth on the design of circumlunar trajectories. (Preprints of a paper for presentation at the January 1961 Inst. of Aero. Sci. meeting were made available to group members at the meeting.)

Calculations (machine and analytic) are underway for the L/D range 0 to 1, W/C_DA range of 25 to 400 psf, and for a load unit of 10g. A mode of operation receiving close attention is a constant coefficient pullup from escape speed, followed by a constant altitude slowup and a constant attitude slowup and a constant coefficient glide. Other modes being considered are a constant aerodynamic landing mode, the return parking-orbit mode (Pritchard) and a skipping mode under the radiation belts. The accessible landing area for vehicles in the parameter range under study is being evaluated. It is hoped to define simple modes of operation for attaining the points of the accessible landing area. For all the trajectories under study, the usual quantities (times, aerodynamic loads, heating rates, heating loads, Reynolds number, lateral and longitudinal ranges) are being evaluated.

A study is being made of the energy requirements for a turnaround performed at various stages of representative circumlunar missions. Reentry angle and velocity and landing point on the earth will be found as a function of velocity increments, the results being presented in a form which allows tradeoff studies to be made. The program is also adaptable to optimization studies of midcourse guidance, where the object is to land in a designated area, not simply to return to the earth. Equations based on the restricted three-body problem of celestial mechanics and spherical trigonometry relationships have been assembled for computation on the IBM-7090 digital computer. Machine calculations will be initiated in the next few weeks.

A study is being made of a range-controlled return of a nonlifting satellite with minimum thrust requirements. The system considers in the most general case the return of a satellite from an inclined circular orbit to a precise target with minimum energy expenditure. A rotating earth is considered and the target need not lie in the plane of the orbit. Instantaneous application of retrothrust and an elliptical trajectory from orbital conditions to reentry conditions are assumed. The case of a polar orbit is considered in detail for several orbital altitudes of 300 to 600 statute miles. Minimum mass fraction, reentry deceleration forces, time of flight, and retrothrust orientation will be determined as functions of target position and longitudinal and lateral range. Computations have been completed for several orbital altitudes and results are being analyzed.

Analytical studies are completed for entry from orbital velocities using altitude control to a reference trajectory and lateral control as a function of heading error. Accurate control is possible to about 85 percent of range capabilities of vehicle.

An analytical study has been completed of range control using a linearized prediction technique. A report is in preparation. This method appears promising and will be extended to include lateral control and entry from parabolic velocity.

The study of guidance of a space vehicle approaching a planet along an entrance corridor reported in NASA TN D-191 is being extended to include use of dead band in guidance logic to reduce corrective velocity required. Results indicate that dead band reduces accuracy with little, if any, savings in fuel. The results will be discussed in a paper for the American Astronautical Society in Dallas, Texas on January 17, 1961. Further plans include study of the effect of instrument accuracies.

A report is in preparation containing analytical studies of penalties associated with simplified techniques for lunar soft landing and return to orbiting vehicle.

Various projection techniques are being studied using a 55-foot diameter inflatable radome as a screen. Simulation of lunar background during lunar circumnavigation is planned.

Calculations are being made of the times of New Moon, First Quarter, Full Moon, and Last Quarter and of the time history of the moon's declination for the period from 1961 to 1971. The results are being assembled in a form which will indicate periods in which the declination is favorable for launching vehicles to arrive at the moon when its surface is under a specified lighting condition. While the results will not provide sufficiently detailed information on the moon's position for trajectory design purposes, they will provide a time basis for interpolating in tables of the moon's coordinates to obtain such information.

Jet Propulsion Laboratory (JPL).-

a. Trajectory analysis.-

Activities at the Jet Propulsion Laboratory relating to Apollo can be divided into two parts, past and current. This note will serve to summarize these activities.

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b. Past activities.-

In executing the Ranger lunar hard-landing, the Mariner interplanetary, and the Surveyor lunar soft-landing programs, JPL has performed, or evaluated, detailed design and analyses of lunar and interplanetary trajectories and midcourse guidance. For the lunar missions, the bulk of the analyses have been concerned with Earth-to-Moon flight only. A very minute amount of time has been devoted to circumlunar studies, reentry, or emergency turnaround characteristics of lunar trajectories.

Detailed launch-to-lunar impact trajectory design and analyses have been accomplished for the Ranger program. As a result of this, the characteristics of lunar-impacting trajectories are well understood at JPL and trajectory design for this type of mission is routine. Reports on results of JPL studies appear bimonthly in the JPL Space Programs Summaries and Research Summaries. In addition, JPL Technical Reports (references 1, 2, and 3) treat trajectory and launch-time problems. The characteristics of the outbound leg of a circumlunar trajectory will most likely closely resemble those of a lunar-impacting trajectory. Thus, techniques developed at JPL may well be useful for the Apollo circumlunar trajectory problem.

In the area of midcourse guidance for lunar impacting and interplanetary missions, a significant amount of design and analysis has been accomplished at JPL. The guidance method employed by JPL for Ranger and Mariner missions involves performing the midcourse maneuvers by radio command from the Goldstone Deep Space Instrumentation Facility (DSIF). The procedure is roughly as follows: Tracking information from the Goldstone, Woomera, and South Africa DSIF stations is transmitted to the JPL computing center in Pasadena, Calif. There, the spacecraft orbit is determined from the data and the correcting midcourse maneuver is computed. The correction is relayed to Goldstone, which in turn, sends the command to the spacecraft. Inherent in this procedure are the analytical computations which must be carried out at the JPL computing center. The basis for these computations has been documented in references 4, 5, 6, and 7.

In the area of trajectory computation, a highly advanced computer program is currently in use at JPL. For lunar trajectories, the effects of the Earth, Moon, Sun, Jupiter, Earth's oblateness, lunar triaxial potential, solar radiation pressure, precession and nutation effects of the equinoxes, etc., are included. In addition to evaluating the position velocity coordinates of the probe, a large number of quantities are computed for engineering design use. The positions of the celestial bodies are obtained from tape-stored ephemeris tables. The method of integration is optionally Cowell or Encke. This program is used operationally in the orbit-determination and midcourse-guidance computations.

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c. Current activities.-

Currently, trajectory analysis at JPL consists of continuation of design for the Mariner missions and precision computation of the Ranger lunar-impacting standard trajectories. Beginning January 2, 1961, the Space Technology Laboratories (STL) started work on a Space Systems Analysis Study Contract which includes a thorough treatment and analysis of circumlunar trajectories and Moon-to-Earth trajectories. This contract is funded to support 2 to 3 trajectory analysts for the calendar year 1961 on these problems. The results of the STL studies will be made readily available to the Apollo program, if desired.

In the area of trajectory computation, a computational procedure called the "Method of Contiguous Conics," is being developed to replace the Cowell and Encke methods. The Contiguous Conics method is basically an extension of the Varicentric method developed by Miner and Sperling of MSFC.

Although little analysis has been done on earth reentry at JPL, a significant effort is currently underway on planetary entry and terminal guidance for the Mariner interplanetary program. The results of these studies may be useful in the Apollo program.

Flight Research Center (FRC).- FRC is doing no work which relates directly to Apollo. These notes report studies which may be of interest to the group.

a. Launch.-

An exploratory investigation is being performed to determine the practicality of air-launching manned vehicles into orbital flight. Although this investigation has been primarily directed to vehicles of the class of DS-1 (10,000-lb weight), the possible use of orbital rendezvous may make it of interest to the Apollo Project (particularly if the Apollo vehicle consisted of several 5,000-lb elements).

Examples of some results of the investigation are:

(1) A two-stage vehicle weighing 200,000 pounds and having an initial thrust-to-weight ratio of 1.5 could orbit 10,000 pounds from a B-52 launch if the vacuum specific impulse of both stages was about 340 seconds.

(2) Use of a B-70 reduces the required specific impulse for the above conditions to about 310 seconds or, with a vacuum specific impulse of 340 seconds, permits orbiting about 13,500 pounds.

(3) If the payload were reduced to about 6,500 pounds, the B-52 could place it in orbit with a two-stage vehicle having a vacuum specific impulse of 310 seconds and weighing about 200,000 pounds.

(4) Storable propellants are probably a necessity, to avoid the topoff problem, although high-energy cryogenic propellants might be considered for the upper stage.

b. Landing.-

(1) X-15 landings.- Results of the first 31 X-15 airplane-approach-and-flare maneuvers have afforded a relatively good cross-section of landing conditions that should be experienced with vehicles of this type. Performance data have shown that the peak lift-drag ratio of the X-15 at approach and landing speeds can vary between 3.5 and 4.5, depending on configuration. These values are somewhat higher than predicted on the basis of wind-tunnel tests. The wing loading is about 65 to 70 pounds per square foot.

Flight data in the approach pattern illustrate the tremendous leeway the pilot has in positioning the airplane for the flare. In performing overhead patterns, the use of speed-brake modulation and maneuvering flight allows the pilot to reduce possible altitude variations of over ±8,000 feet on the downwind leg of the pattern to touchdown dispersions of the order of ±1,000 feet.

Most problems encountered in landing the X-15 were experienced during the flare, primarily because of the severe limitations placed on touchdown angle of attack. The average flare commences at an altitude of about 800 feet and a speed of 300 KIAS. When the flare is essentially completed, the flaps and gear are extended. Touchdown is then accomplished at about 185 KIAS with an angle of attack of approximately 7° and a rate of sink of about 4 fps.

All the extensive preflight predictions have essentially been verified in flight. An F-104 which has been modified to simulate the X-15 is proving invaluable in pilot training for X-15 landings.

(2) DS-1 landing simulation.- Because of the gratifying results obtained with the F-104A in simulating in flight the X-15 landing, a similar program is now underway for the DS-1. An F5D airplane will be used for these tests because it closely approximates the DS lift-drag-ratio characteristics and wing loading. The pattern and the flare will be examined. Because the DS-1 will, on occasion, perform the approach through cloud cover, the landing-aid requirements will be studied.

Patterns will be flown with and without guidance to aid in establishing optimum pattern geometry. Consideration will be given to

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airplane visibility effects on the pilot's ability to perform the pattern approach.

In the flare an attempt will be made to establish the values of the various flare-control parameters as well as touchdown conditions, including touchdown dispersions.

(3) Landing of a lenticular vehicle.- The landing characteristics of a vehicle of circular plan form having a lenticular cross-section are being investigated. The analytical investigation of a vehicle having a wing loading of 38 pounds per square foot indicates that the pilot should have no trouble executing a normal approach and flare for a touchdown at little or no vertical velocity. An investigation is in progress of the possibility of simulating the landing of the lenticular vehicle with a current fighter.

c. Abort.-

DS-1 abort and escape.- The primary escape mode provided the Dyna-Soar pilot consists of glider-abort separation from the boosters. A subsonic ejection seat is provided for pilot escape from the glider in instances when a satisfactory landing site cannot be reached or when other conditions make an attempted glider landing impractical.

Glider abort can be initiated from the launch pad or at any time up to second-stage booster separation by the pilot or by automatic sensing devices. After second-stage burnout, all escape implementation is the sole responsibility of the pilot.

In the event of fire or explosion among the booster or pad installations after the pilot has boarded the glider, or of flight malfunction dictating a need for immediately separating the manned glider from the booster, the pilot can initiate ignition of the glider separation rocket. If this abort action is taken from rest on the pad, the glider will be maneuvered to make a west-to-east approach and landing on the Cape Canaveral skid strip. If the abort occurs after lift-off, the pilot will guide the glider to an optimum landing site chosen on the basis of velocity, altitude, and position at termination of the separation rocket thrust. A study of the required down-range landing sites is currently underway. Any emergency landings will be made at down-range fields existing primarily for normal logistics traffic. The Flight Test Control Center and its down-range extensions (Station Monitors and Landing Controllers) will advise the pilot in the selection of the emergency landing site and will furnish steering direction as requested by the pilot.

A brief, analytical investigation has been made at FRC of the practicability of executing the off-the-pad maneuver using the planned

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abort rocket. In addition, the pilot's ability to recover the vehicle following boost-off was evaluated with an F-100 airplane simulating the flight conditions for the escape.

The escape rocket was assumed to place the vehicle in vertical flight at an altitude of 2,700 feet and a speed of 1,050 fps (620 knots). Following a pitch-roll maneuver to attain level flight, the vehicle was then glided to a nearby emergency landing strip. The results of the investigation indicate that the recovery as proposed is within the capabilities of both the pilot and the vehicle. Because of altitude restrictions, it was not feasible, with the F-100, to simulate completely the glide and landing phases of the recovery. It is planned to continue this investigation with a more suitable airplane and to extend the analytic investigation to lower altitudes and lift-drag ratios.

Lewis Research Center (LeRC).-

a. Lunar trajectory study.-

The objective is to obtain sufficient data on trajectories to permit the carrying out of manned-lunar-mission analyses, with realistic consideration of launch times and dates, selection of favorable launch sites, comparison of direct versus orbital departure, etc.

The purpose of this study is to analyze three-dimensional, simplified two-body coast trajectories between the earth and the moon, with some consideration of the interactions with earth and lunar takeoffs, and to calculate relationships between ΔV requirements, flight time, location of launch site, etc.

Curves have been drawn of the various angular relationships, flight time, burnout velocity at start of trajectory, typical ΔV to decelerate at the moon, and so forth, as functions of burnout longitude and latitude and position of the moon. The above-described data are being incorporated in a NASA Technical Note, now in the editorial process.

b. N-Body trajectory program.-

The objective is to provide a general, precise, rapid, and flexible machine program for the solution of trajectory problems related to various space missions. The program shall be capable of solving such problems as: boost from a planet surface, precise lunar trajectories, including effects of earth oblateness and solar and lunar perturbations, and interplanetary trajectories including the perturbations of the various planets in the solar system.

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A general FORTRAN coded-machine program for the precise solution of the N-Body problem of space mechanics has been developed. The equations of motion are written to include: (1) no more than 8 celestial bodies, (2) the effects of oblateness and atmospheric forces from the celestial body currently serving as the coordinate origin, and (3) propulsive forces.

Major features of the existing program are:

(1) A fourth-order Runge-Kutta integration process with automatic step-size control and double-precision accumulation of the variables of integration.

(2) A choice of integration in either of two coordinate systems is provided. These are: conic section orbit elements and Cartesian coordinates (Cowell's method). If the former method is chosen, provision is made for automatic temporary conversion to Cowell's method to avoid difficulties which occur when eccentricity is very close to zero or unity, or when the vehicle is close to the asymptote of a hyperbolic orbit.

(3) Precise planetary position and velocity data are obtained from fifth-order polynomial fits of ephemerides. The coefficients of the polynomials are simultaneously fitted to both position and velocity at three dates. The dates are selected such that the greatest time interval, consistent with seven-digit-position accuracy and six-digit-velocity accuracy, is available.

(4) For exploratory work not requiring great precision, approximate ephemerides can be obtained by a built-in iterative solution of Kepler's equation with assigned planetary orbit elements.

(5) Automatic transfer of the origin of coordinates is provided to permit the largest integration intervals consistent with the desired precision.

The program has been used to provide checks on various lunar and interplanetary trajectories obtained by more approximate techniques.

c. Boost trajectories.-

A set of working charts relating the optimum-stage propellant-fractions for two- and three-stage rocket vehicles for launching payloads into nominal low-altitude circular orbits about the earth has been developed. The charts contain data for various combinations of stages using RP-Lox and H_2-Lox-propellant combinations. These specific results can be extended easily with little error to other propellant combinations.

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Together with stage-structural and propulsion-system weights, the charts are useful for preliminary design optimization of boost vehicles.

Two different computer programs were used to calculate the data for the charts. First-stage trajectories are gravity turns and were numerically integrated and include the effects of aerodynamic drag and the variation of engine thrust with altitude. Second- and third-stage trajectories were calculated using approximate closed form solutions of the calculus-of-variations trajectories (i.e. "linear-tangent" thrust attitude). This second program contains an iteration scheme which permits the determination of the top-stage propellant fraction for specified terminal attitude, velocity, and flight-path angle.

These data are reported in a report "Performance Charts for Multi-Stage Rocket Boosters" by MacKay and Weber (NASA TN D-582). This report has been transmitted to NASA Headquarters and should be issued shortly.

d. Atmospheric entry and braking.-

A three-part study of the regimes of atmospheric flight applicable to space missions is in progress. The analysis is broad in scope and is formulated to yield closed-form approximations to the solutions of the equations of motion and heating. From such approximate solutions, it is possible to survey quickly the entire realm of entry paths, etc. and to delineate the more important design and operating variables.

The three parts of the study and the corresponding simplifying assumptions which permit the closed-form solutions are:

(1) Entry corridors and flight-path angles

$$\cos\psi \approx 1.0 \qquad (\psi = \text{flight-path angle})$$

$$\overline{V} = \text{constant}$$

(2) Motion and heating along flight paths for deceleration from supercircular speeds

$$g \sin\psi << \frac{1}{2} C_D \rho V^2 A$$

$$\cos\psi \approx 1.0$$

$$d\psi/dt \neq 0$$

$$\rho = \rho(\overline{V})$$

(3) Motion and heating during atmospheric passes

$$g \sin \psi << \frac{1}{2} C_{D0} V^2 A$$

$$\cos \psi \approx 1.0$$

constant flight-path radius

The validity of these approximations has been checked with the results of machine integrations reported by Chapman and Becker. The closed-form solutions yield results sufficiently accurate for preliminary design survey purposes.

Each of the studies covers cases with modulated as well as constant angle of attack. The results show the effects of maximum g load, initial velocity and configuration maximum and operating L/D. The solution of the equations of motion are applicable to any planet; those for the heating are applicable to any planet with an atmosphere with heating characteristics similar to those of earth.

The first study has been completed and a report "Approximate Analysis of Atmospheric Entry Corridors and Angles" by Roger W. Luidens (NASA TN D-590) has been transmitted to NASA Headquarters for final approval. In this report the modes of vehicle operation giving the deepest corridors are defined. The effects of hot gas radiation and a Reynolds number limit on corridor depth are discussed.

In part 2 of the study, six flight paths that are considered appropriate for decelerating from supercircular speeds after an entry are analyzed. Among the paths studied are those characterized by: (1) constant angle of attack, (2) constant Reynolds number, and (3) constant g. For vehicles with laminar boundary layer, relations are developed which yield the heat rate, the heat input per unit area during the deceleration, and the total heat input to the vehicle. Of the flight paths considered, the constant g path (with g equal to the g limit) yields the lowest total heat input, about half that for a constant angle-of-attack path with the same initial values of velocity and g. A report has been prepared on the study and is in the editorial process.

The third part presents the velocity change and heating incurred during a pass through the atmosphere as functions of such variables as maximum g load, initial velocity, L/D, and drag and lift-modulation ratio. This study includes the concept of the variable-area drag chute. Work on this phase is in progress.

e. Guidance studies.-

The problem of midcourse and approach (terminal) guidance for interplanetary missions is under study at the LeRC. The objective of the study is the formulation and evaluation of guidance concepts which may be employed to prescribe efficient trajectory control. Consideration has been given primarily to passive-measurement systems (optical, infrared, inertial), and the performance of components was postulated in functional form and the effects on guidance evaluated over a wide range of error assumptions. The feasibility of accurate guidance with minimum propellant expenditure has been demonstrated for components of modest performance capability, however, multiple impulsive corrections are required. Some characteristics of the study are indicated in the following paragraphs.

Approach guidance.- That portion of the transfer trajectory from about the sphere of influence of the target planet down to the sensible atmosphere is considered the realm of approach guidance, and has been treated in references 8 to 11. A two-body, two-dimensional analysis with the perigee distance serving as the target parameter has been assumed. Reference 8 presents a documentation of ΔV requirements to correct nominal trajectory errors, and shows that circumferentially-applied thrust represents a very good approximation to the optimum thrust direction. Some preliminary studies of a statistical nature are presented in references 9 and 10 for two distinct measurement schemes. These early studies led to a refinement and integration of guidance techniques and methods of evaluation, and a fairly complete digital computer guidance program was evolved. The analysis and statistical performance evaluation are presented in reference 11, soon to be published. Nondimensional results applicable to any target planet are shown for characteristic solutions and for extensive parametric studies of variations in trajectory and measurement error parameters. The use of least-squares data adjustment and guidance logic techniques has proven very satisfactory as a means of improving the efficiency of guidance maneuvers. Attainment of entry corridors on the order of 10 miles has been demonstrated for measurement errors up to 2 minutes of arc RMS. The ΔV cost, of course, is strongly dependent on the accuracy of midcourse guidance.

Midcourse guidance.- A program to study correctional maneuvers during the midcourse phase of a mission has been developed and evaluated. The fundamental guidance theory is based on well-known techniques of linear perturbation methods. Specifically, angle measurements are used to indicate position deviations from a precomputed reference trajectory, which in turn yield the amount of velocity correction to be applied. A two-body, three-dimensional analysis is assumed. The concepts of data adjustment and logic techniques are also included in the guidance theory, and are found to have a significant effect on the accuracy and ΔV cost of corrective maneuvers. Unpublished results of a typical Mars mission evaluation show that a vehicle using instrumentation accurate to 40 seconds of arc RMS can be guided to within 4,500 miles of the planet center for a total ΔV expenditure of less than 360 ft/sec. An average of five and a maximum of nine corrections are required.

The results of the midcourse and approach guidance studies can be integrated to yield overall guidance requirements. It appears feasible to guide a vehicle to within a 5-mile entry corridor for a total ΔV expenditure of under 1,000 ft/sec. The necessity of multiple corrections, however, is reiterated.

Areas requiring further study.- It was generally agreed by the group that there was an urgent requirement to develop a standard model for the Van Allen radiation belt which would be used in all trajectory analyses related to Apollo.

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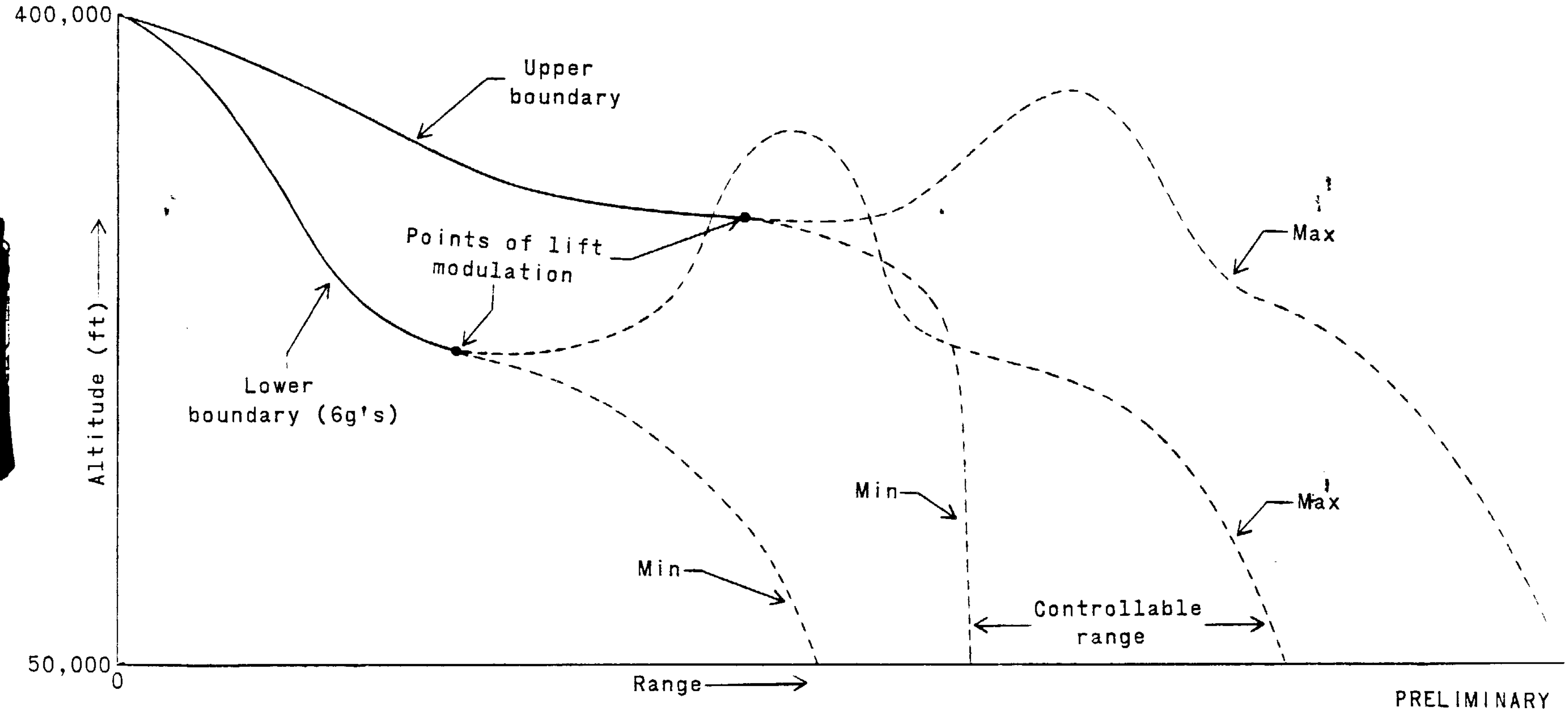
[redacted]



Section I, Figure 1.- Illustration of controllable range.

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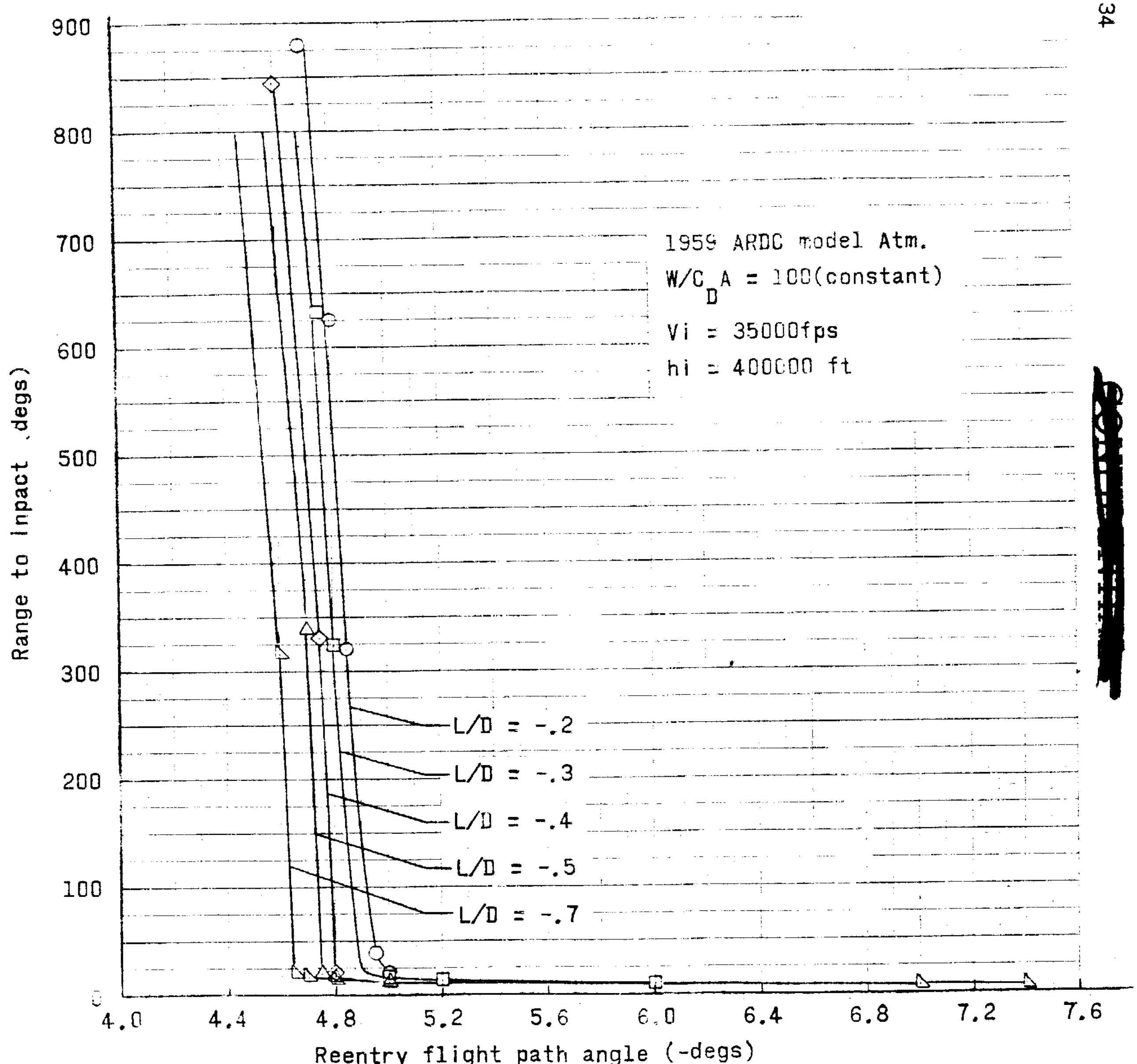

PRELIMINARY

Section I, Figure 2.- Range near the top of the corridor.

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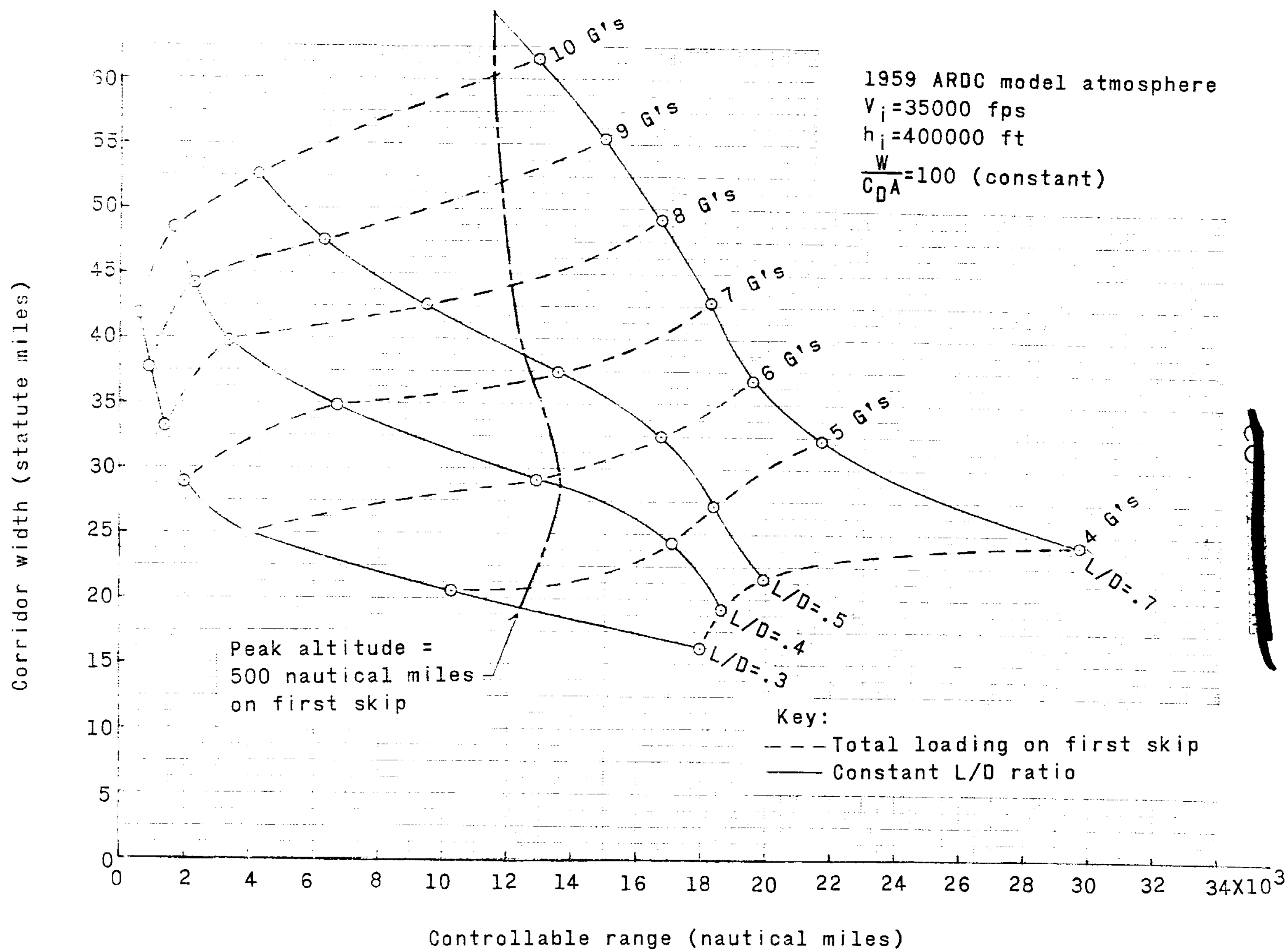
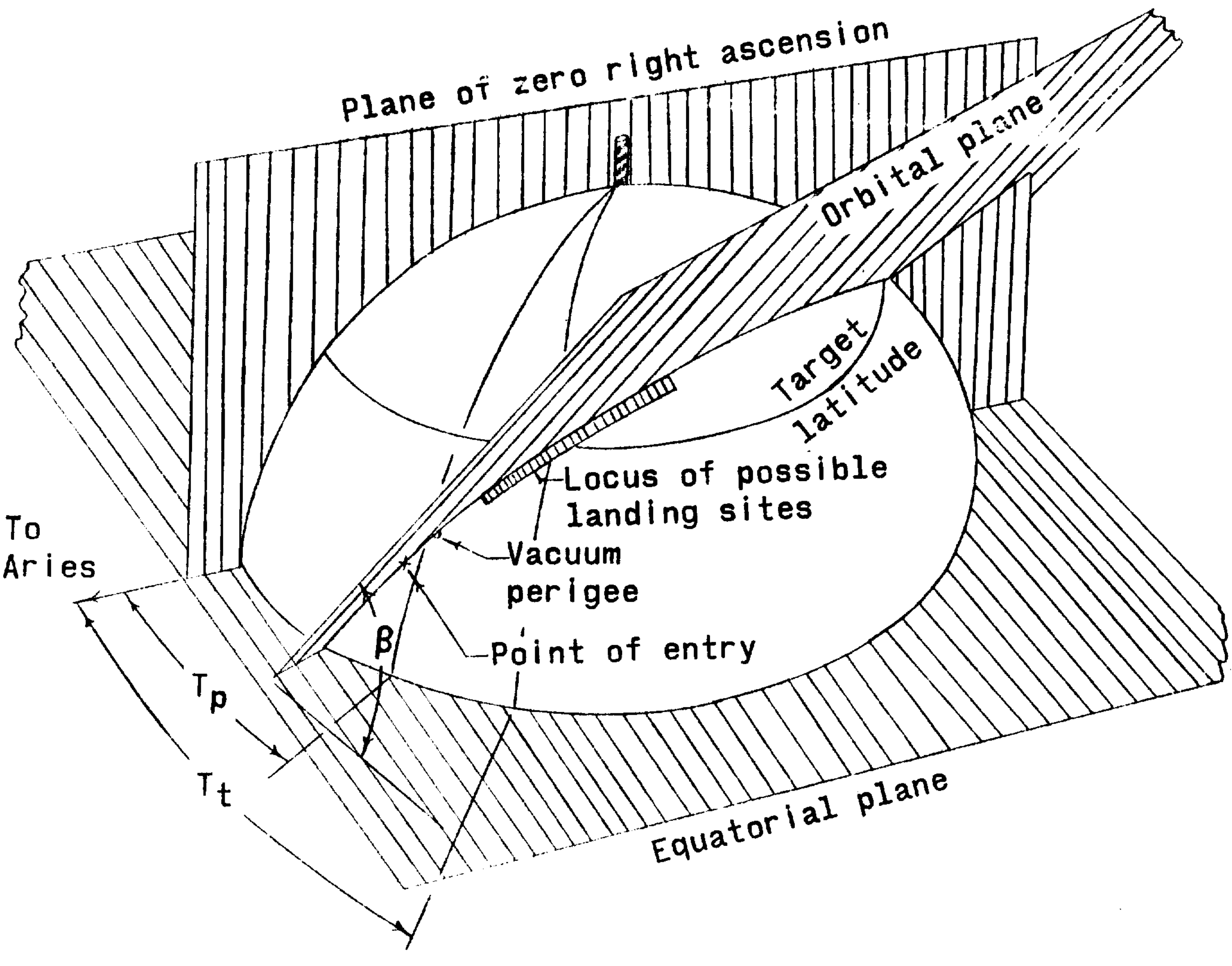


Section I, Figure 3.- Controllable longitudinal range of lifting vehicles for-reentry near escape speeds.



Section I, Figure 4.- Geometry of reentry.

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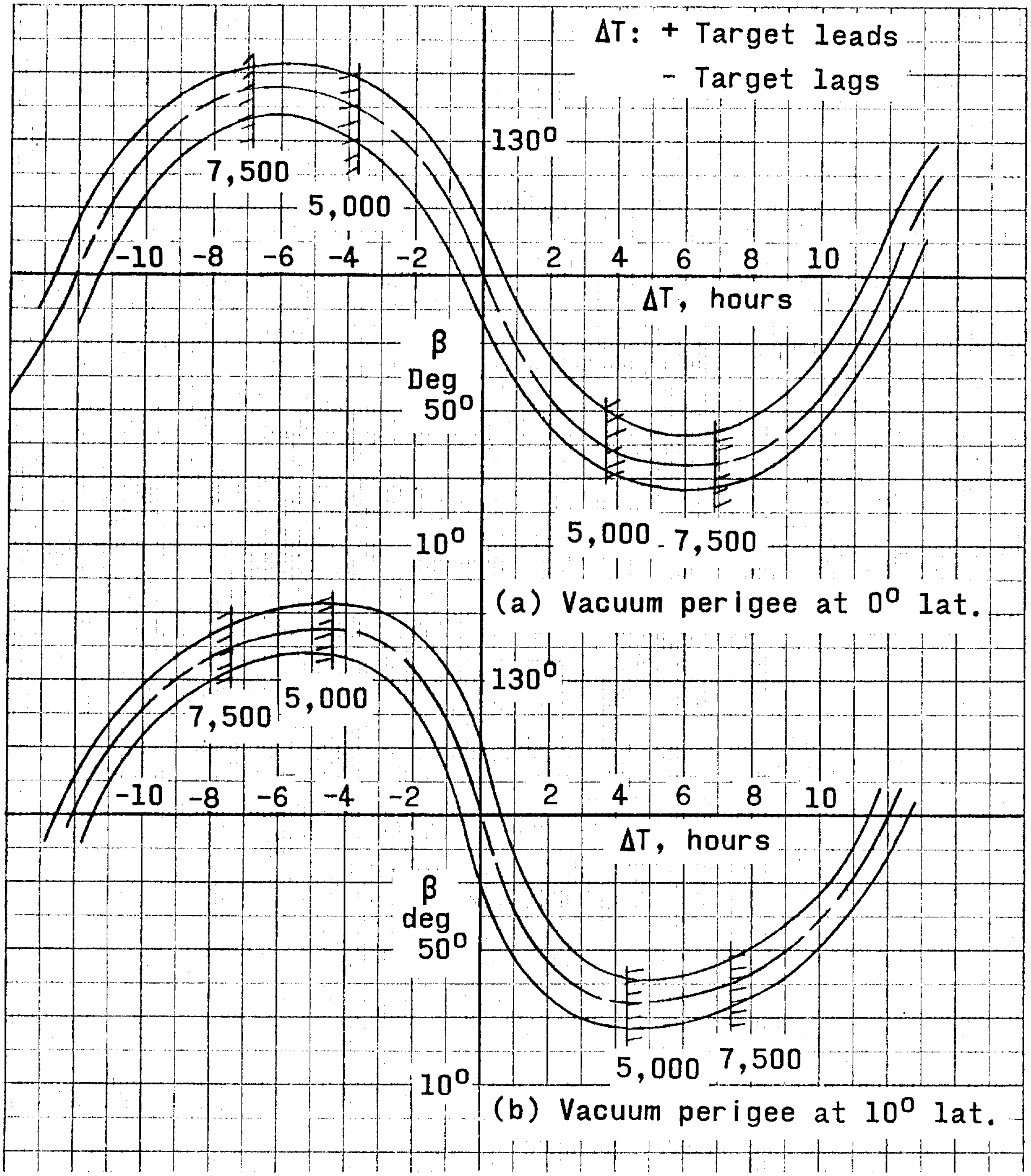

Target latitude 35°

Section I, Figure 5.- Variation of obrital plane inclination and relative longitude of perigee and target.

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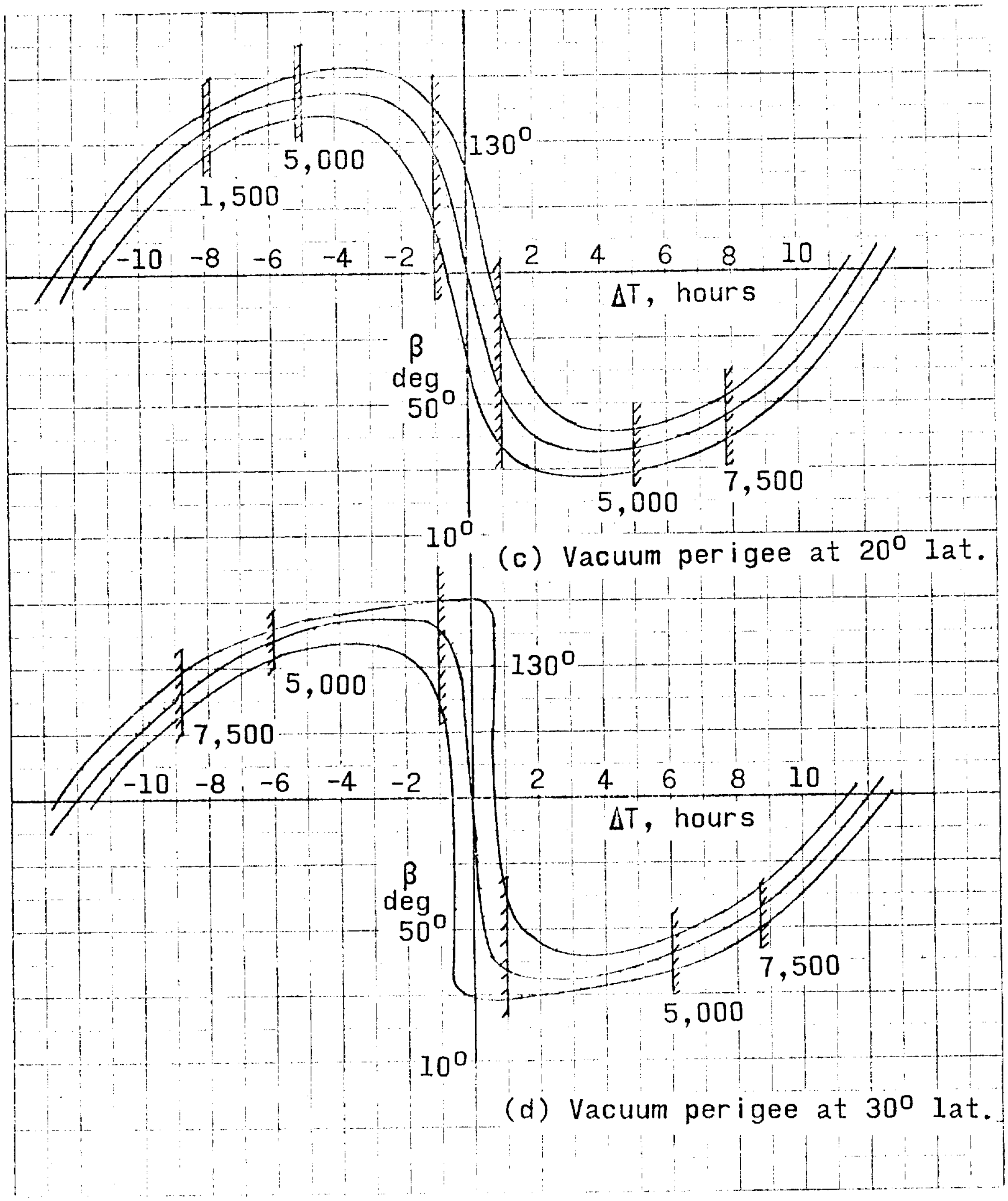

(c) Vacuum perigee at 20° lat.

(d) Vacuum perigee at 30° lat.

Target latitude 35°

Section I, Figure 5.- Concluded.

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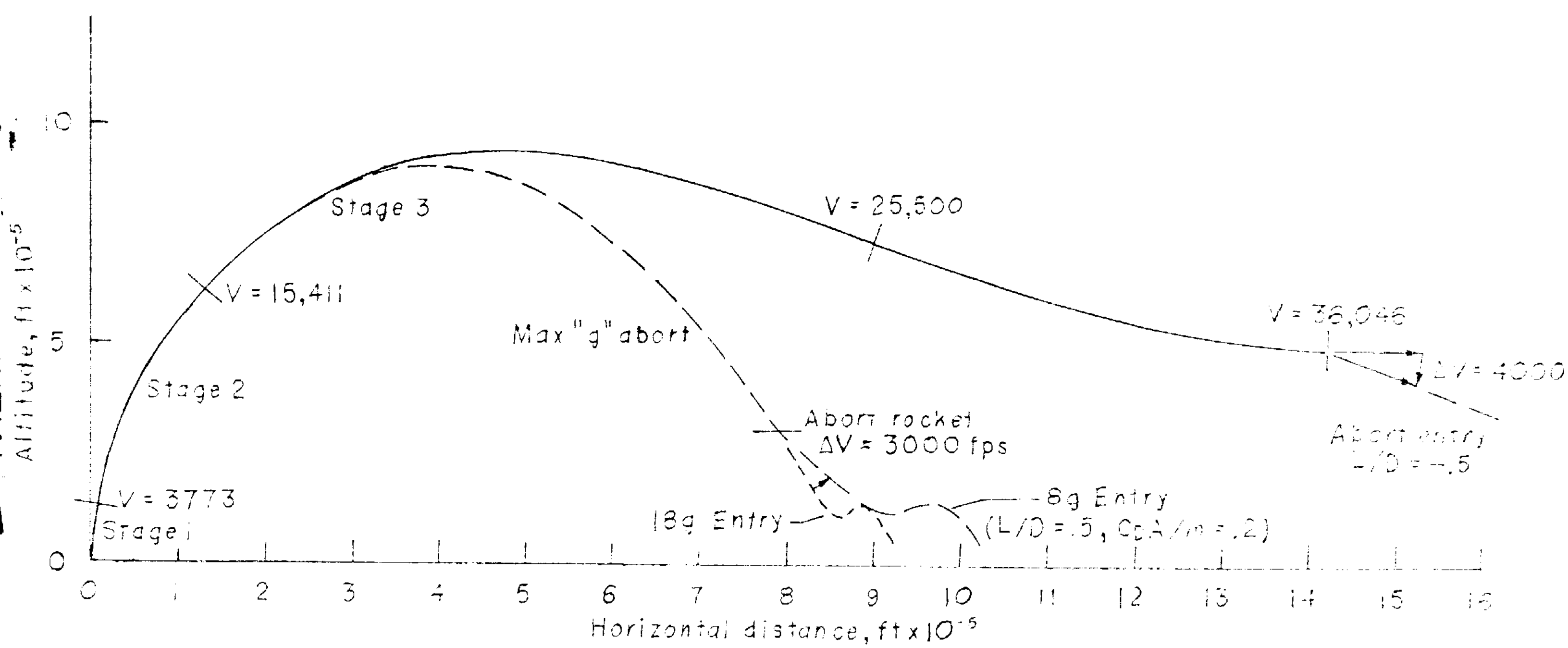

Section I, Figure 6.- Typical three-stage boost trajectory.

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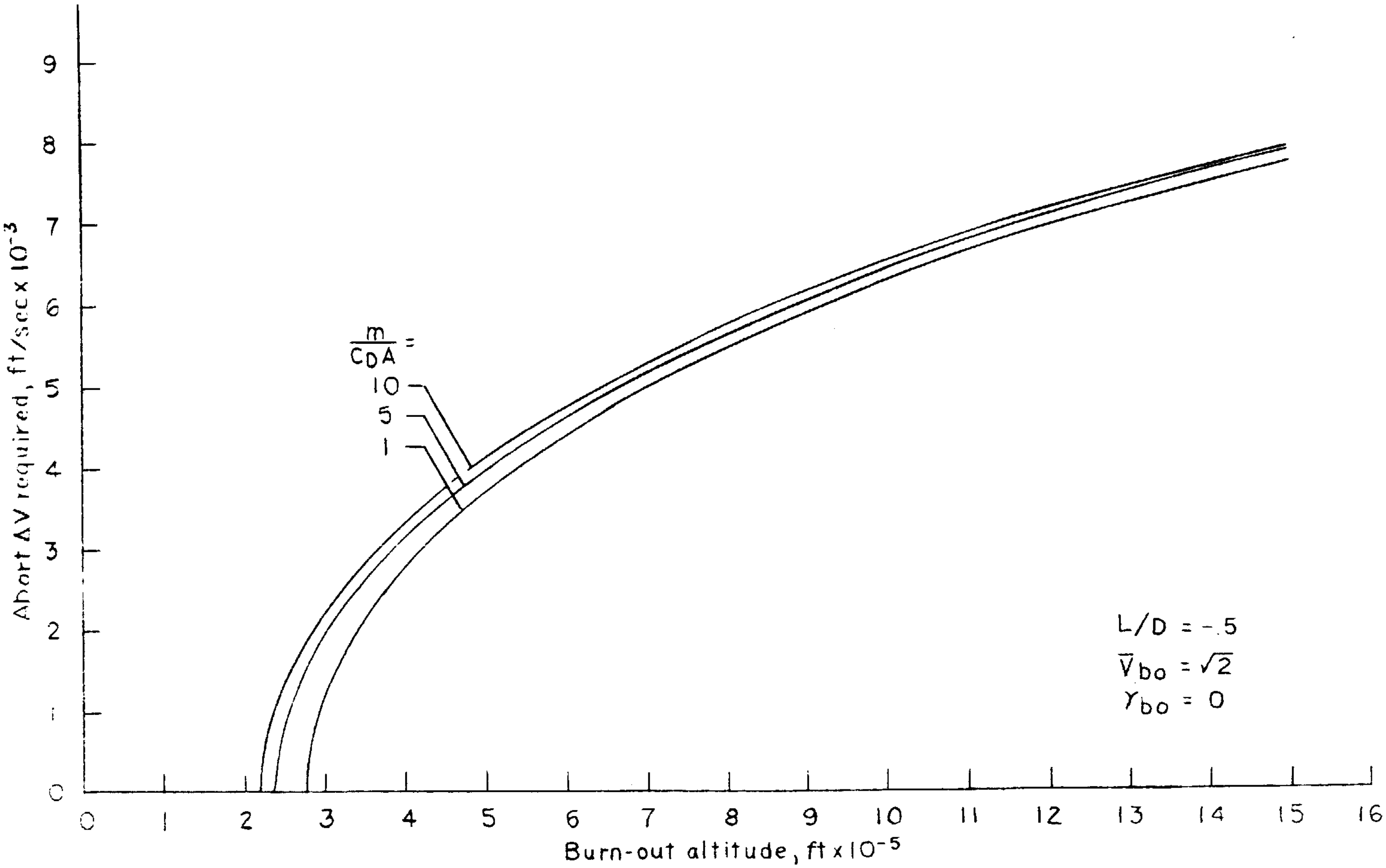

Section I, Figure 7.- The effect of vehicle $\frac{M}{C_D A}$ on abort ΔV requirements.

[redacted]

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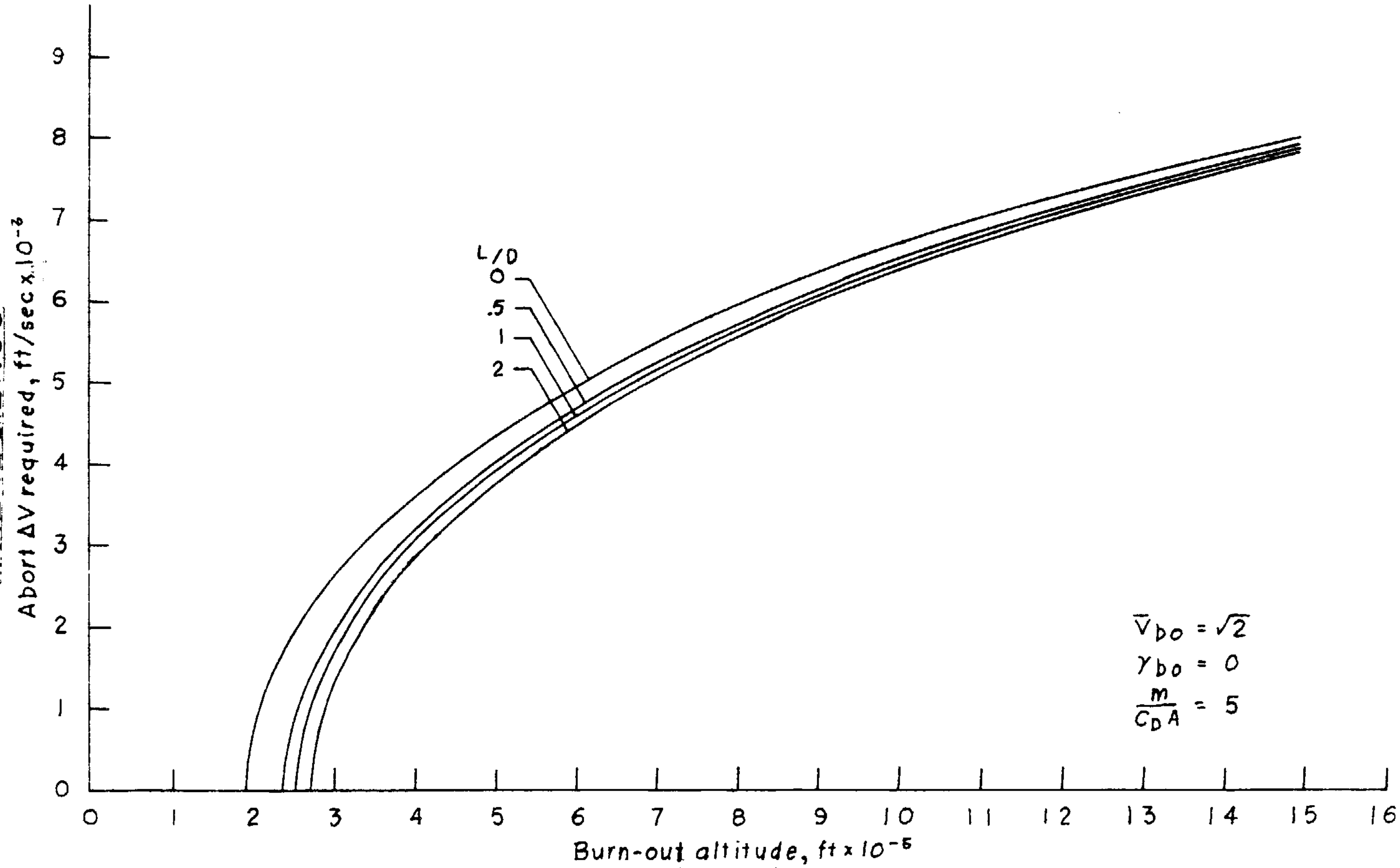

Section I, Figure 8.- The effect of vehicle L/D on abort ΔV requirements.

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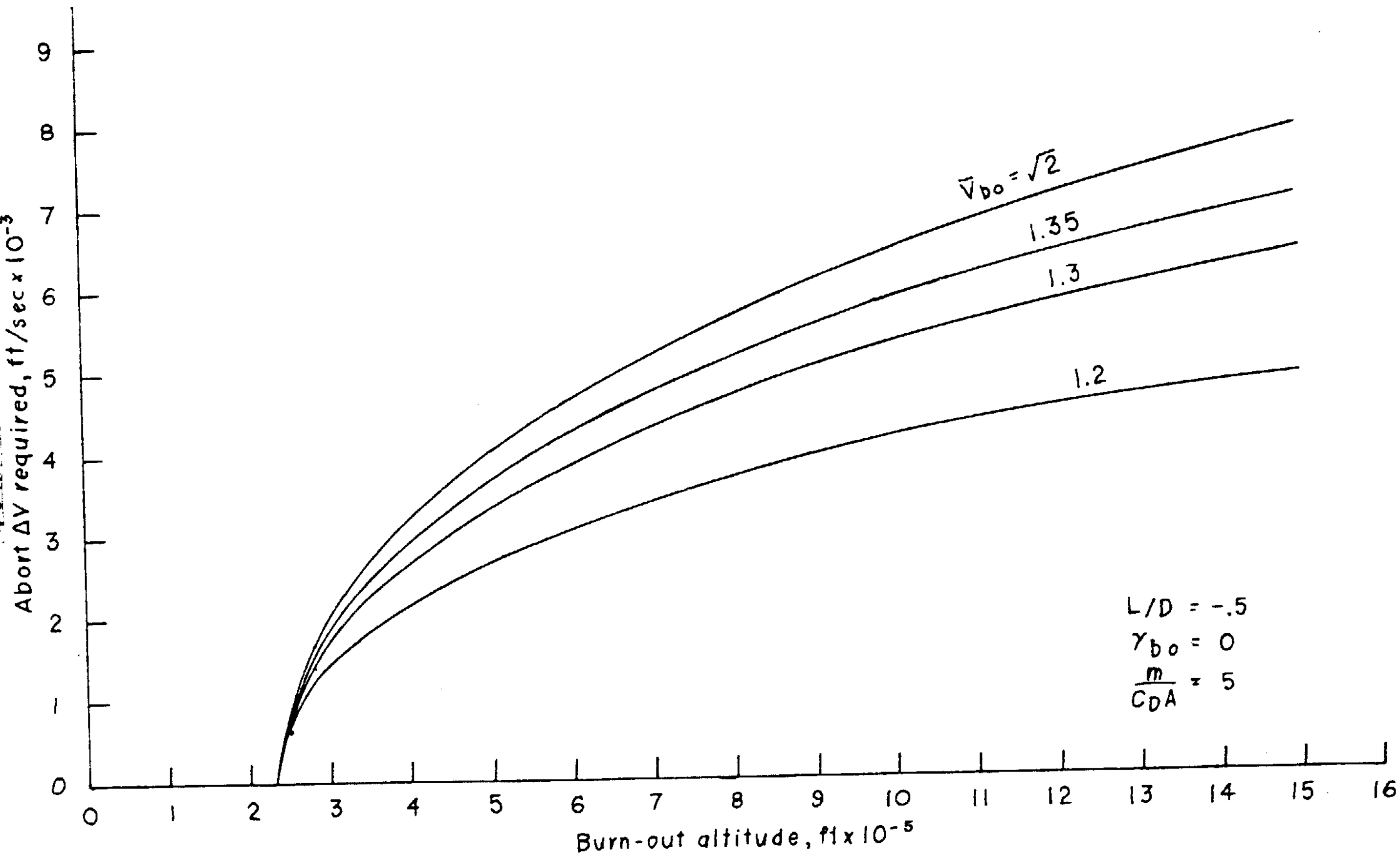
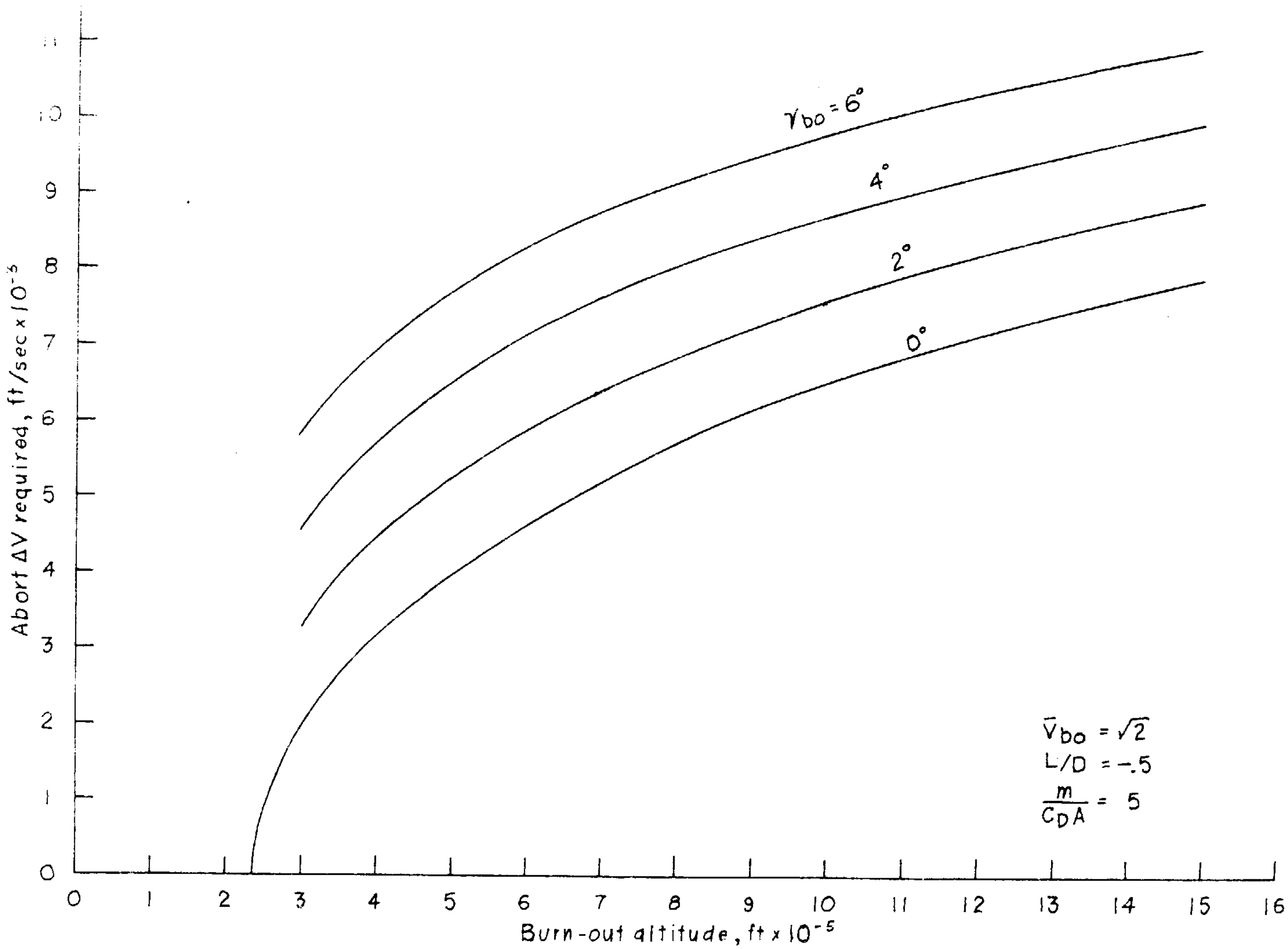


Section I, Figure 9.- The effect of burnout velocity on abort ΔV requirements.

[redacted]

[redacted]



Section I, Figure 10.- The effect of burnout path angle on abort ΔV requirements.

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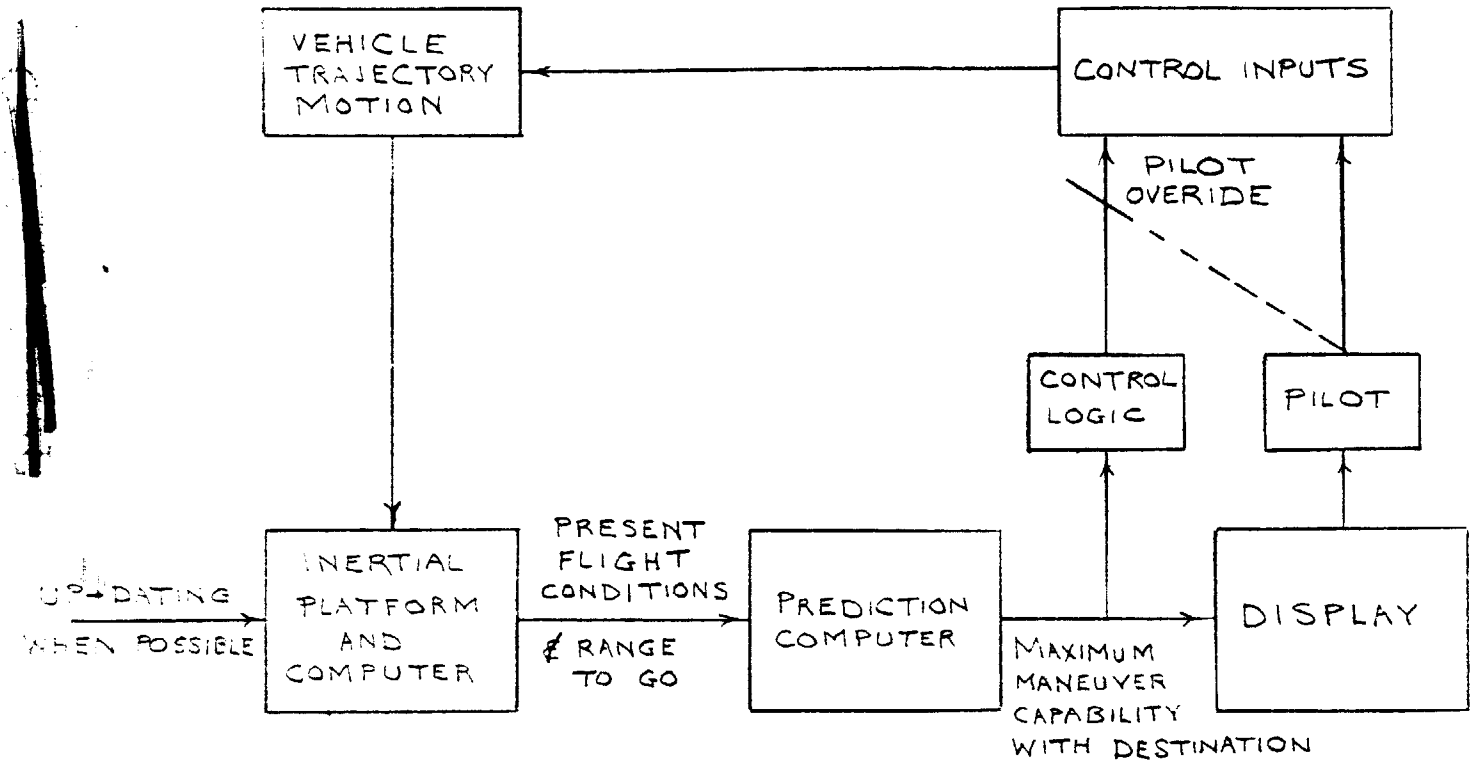
[redacted]



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Section I, Figure 11.- Proposed navigation concept.

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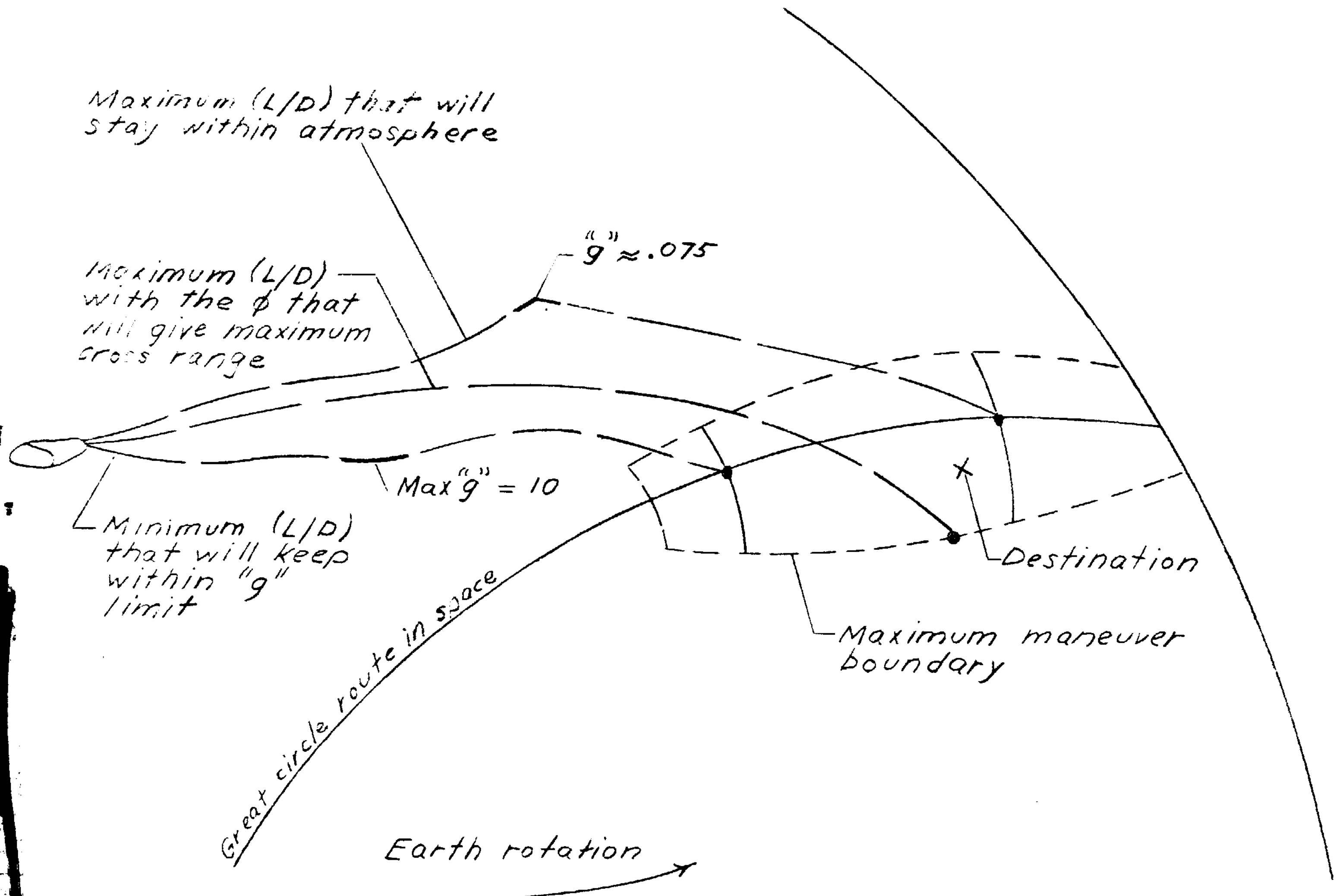

Section I, Figure 12.- Method of predicting the maximum maneuver capability for the lunar mission reentries.

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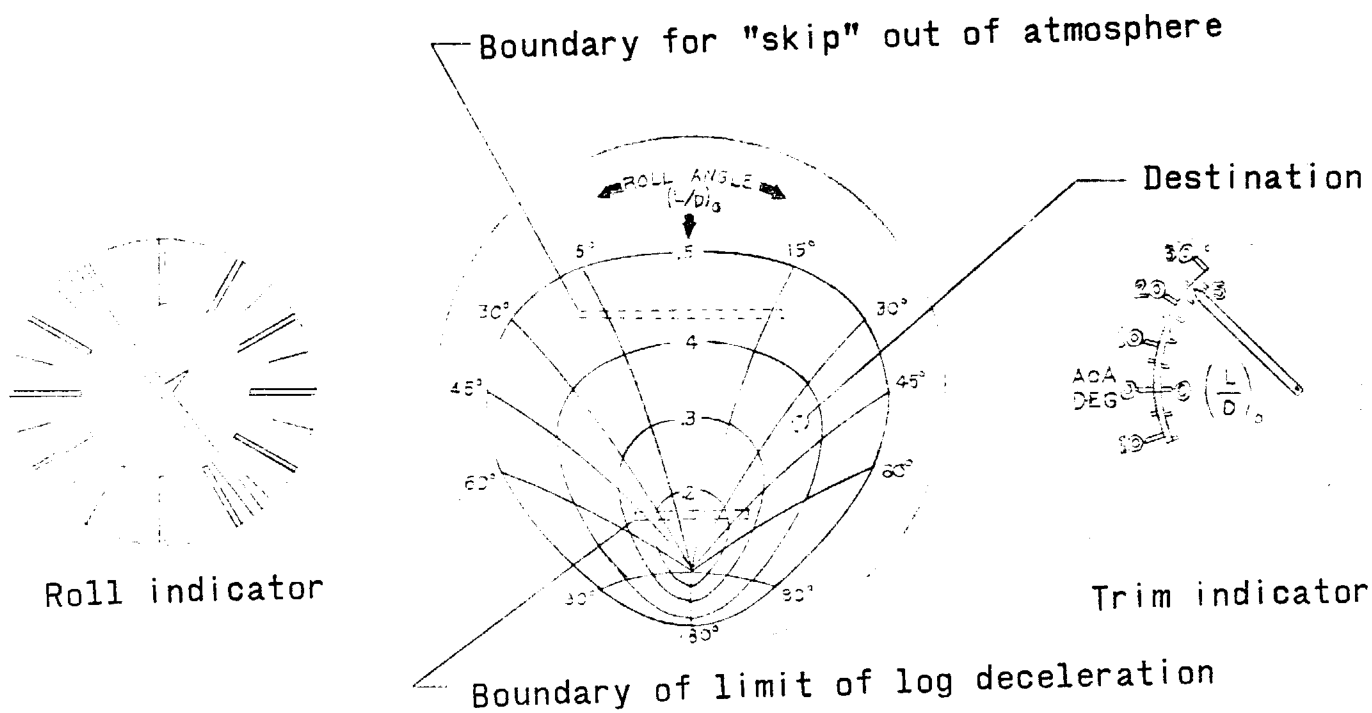

Section I, Figure 13.- Guidance and control display.

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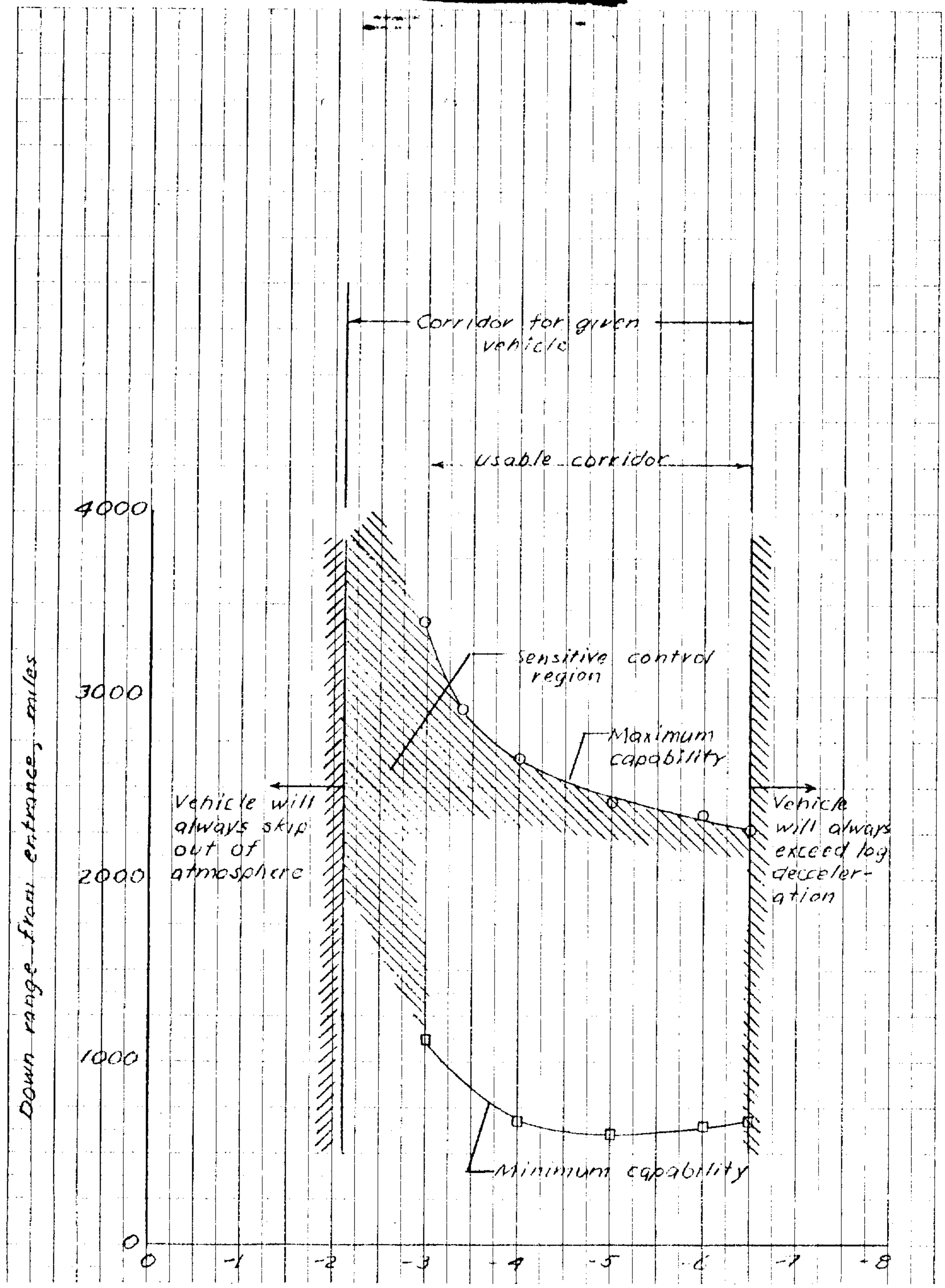

Entrance flight path angle, degrees

Figure 14.- Down-range and corridor boundaries for the entrance condition of h = 290,000 feet and V = 35,000 feet per second $(40)_{max}$ = 0.5.

[redacted]

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SECTION II

EXCERPTS FROM

INSTRUMENTATION AND COMMUNICATION

APOLLO TECHNICAL LIAISON GROUP MEETING

January 6, 1961

Space Task Group

Langley Field, Virginia

[redacted]

Reports from various Center members on Apollo-related activities were:

a. Space Task Group (STG).- Studies presently being conducted and reported are:

(1) Preliminary Concept for Communications and Tracking Equipment

(2) Preliminary Investigation of a Television System for Use on the Apollo Spacecraft

(3) Compilation of Worldwide Tracking and Communications Facilities Offering Possibilities for Use in Project Apollo

(4) Preliminary Consideration of Radio Beacon System for Recovery of Apollo Spacecraft

(5) Preliminary Feasibility Study of a PCM Telemetry Link for Project Apollo

(6) Preliminary Investigation of Ground Tracking and Communications System Adaptable for Use in Project Apollo

b. Marshall Space Flight Center (MSFC).- A brief account of Instrumentation and Communications Branch work at MSFC was given. The branch is broken into the following groups:

(1) R. F. System Section

(2) Measurement Section

(3) Telemetry Section

(4) Planning Section

(5) Special Projects Office

Reported and discussed briefly were the following areas of work presently being studied at MSFC:

(1) SA-4 Instrumentation Plan

(2) Saturn Frequencies

(3) Horizon Sensors

(4) Radar Altimeters

(5) Television

c. Langley Research Center (LRC).- Areas of interest that would be Apollo-related and are under current study by LRC are:

(1) Propagation Through Ion-Sheath

(2) Antennas

(3) Reflectometer

(4) Radar Altimeter Use in Moon's Proximity

(5) Radiation Effects on Solid State Materials

A discussion of areas requiring further study and areas where studies should be initiated resulted in the following list:

a. Ion-sheath problem for reentry

b. Radar-altimeter

c. Antennas (general)

d. PCM transmitting system

e. Television

f. Onboard data recording (sensors and storage)

g. Data reduction

h. Radiation sensors

i. Bioinstrumentation

j. Measurement requirements

k. Frequencies

l. Beacons

m. Recovery aids

[redacted]

SECTION III

EXCERPTS FROM

PROJECT APOLLO MINUTES OF MEETING

OF

MECHANICAL SYSTEMS TECHNICAL LIAISON GROUPS

January 6, 1961

Space Task Group

Langley Field, Virginia

[redacted]

Space Task Group (STG) gave a brief discussion of possible requirement for the provision of artificial gravity in the Apollo system, and requested that panel members consider the effects such a requirement might have on the various mechanical systems.

STG in-house studies on the mechanical systems for Apollo:

Environmental Control System.- This study has been confined to the thermal control to date. An indirect heat transfer system was assumed to simplify the magnitude of this study and has been shown to be compatible with the surface area available for the heat load of 7,620 Btu/hr. The radiator surface versus tube spacing and skin conductivity, thickness, and surface conditions have been parametrically studied.

Reaction Control System.- The most feasible system appears to be one using storable hypergolic propellants. Since the fuel consumption varies as the square of the minimum repeatable pulse, Marquardt's PAT-C system is of interest. They have offered NASA the necessary hardware on consignment for an evaluation program and Lewis Research Center will conduct the program. The first unit will be a 25-pound thruster with two solenoids and a wave shaper. It is hoped that this program will continue into an evaluation of a single driving unit. It was stated that a preliminary estimate of the total impulse required during the mission for attitude control was 100,000 lb-sec. Propellant pressurization would employ bladders and cold gas.

Similar systems have been discussed with RMD of Thiokol Chemical Corp. and Bell Aircraft Corp.

Auxiliary power supply.- A first look at the mission power requirements indicates that the average load for the major portion of the mission will be 1,800 watts. This is expected to increase after further studies have been completed on all systems. A fairly extensive literature search has been done and is continuing on a preferential basis. As a result of this, a first cut indicates that solar turboelectric solar cells plus NiCad batteries and the cryogenic reciprocating engine should be considered. (See fig. 1.) It is suspected that the data used for the reciprocating engine may be too optimistic; however, Vickers is in the process of certifying these data to the extent possible. Similarly, it is suspected that the data on the fuel cell are pessimistic since the fuel consumption rate of the two is nearly the same. Additional weight requirements imposed on the Reaction Control System also have not been assessed for the solar systems.

In addition to Vickers, STG has talked to Marquardt in terms of what could be done with their hydrazine-nitrogen tetroxide engine with cryogenics.

With solar cells, there seems to be considerable differences of opinion as to the percent charge versus cycle life of secondary batteries.

Present plans are to review the weights of the most feasible systems and choose one as a main supply for the in-house study with provisions for emergency power for the minimum requirements.

Langley Research Center (LRC) Apollo work to date.-

Auxiliary space power.- The manned circumlunar mission has been studied at LRC in an effort to define the type of power system which would be most applicable and to uncover research problems. Ground rules assumed that the mission length would be 7 days, although consideration was given to the probability that provision would have to be made for an extra 7 days resulting from initial guidance errors. A three-man crew was assumed and a separate reentry capsule with its own power supply would be used rather than having the complete vehicle return to earth.

A survey of power consuming apparatus which would be onboard was made along with estimated power consumption at all times during the mission. Without making a detailed breakdown, the results of this survey are given in Table I. A summation of the load patterns of these items was made and is reproduced as figure 2. If solar power is considered for the mission, the possible dark periods are of interest and are indicated at the bottom of figure 2. They include an hour in the earth's shadow at launch, an hour behind the moon (at 60 hrs), and 4 $\frac{1}{2}$-hour periods when course correction rockets are fired and the solar power apparatus may for a time have to be oriented in a direction other than toward the sun.

There are several possible choices for a power supply for this mission. These include both power systems (solar or nuclear) and energy systems (chemical). SNAP-2, a 3kw reactor power supply, will be available at a weight of about 500 pounds without shielding. The shielding has been estimated at 1,500 pounds for use on a manned mission, however, making the total weight rather high. The Sunflower I 3kw solar power system will be available at around 600 pounds. A possible objection to this system on a small vehicle is that movement by the occupants of the vehicle will cause the stabilization and orientation system to be working continuously to hold the mirror alinement with the sun to within about $1/3^{o}$. The penalty involved in this requirement has not been evaluated to date. The use of solar cells would reduce this difficulty, since they are much less sensitive to orientation. For the dark periods, the Sunflower I incorporates thermal storage, but for a solar cell array, batteries would have to be provided. It can be shown that, for this purpose, primary batteries should be used rather than rechargeable storage batteries. Estimated weight for such a solar cell-primary battery system is 614 pounds.

Chemical systems may also be considered, although they are high on weight for missions as long as 7 or 14 days. Figure 3 shows some estimated possibilities for this mission. One possibility which has not been fully explored is to use, for the solar flare and other radiation shielding of the vehicle, a material which may also be used as fuel in a chemical power system. In this way, not all of the fuel weight would have to be charged to the power system. Some limited study of this arrangement is under way at LRC at the present time.

TABLE I

ESTIMATED POWER REQUIREMENTS

Item	Power, watts Peak	Minimum
Controls	100	0
Guidance and Navigation	975	450
Midcourse Guidance	800	0
Communication	400	30
Flight Test Instrumentation	682	682
Scientific Payload	300	0
Life Support	1,200	1,200

It was pointed out that LRC weight estimates for auxiliary power systems are somewhat at variance with those of STG. It was also noted that an additional weight penalty involved in the use of Sunflower would be the nose cone for launch and exit required to cover the folded reflector.

LRC is of the opinion that silicon solar cells are preferable to Sunflower for Apollo since the solar cell angular alinement required is about $\pm 8^{\circ}$ to 10° as compared to $\pm\frac{1}{3}^{\circ}$ for Sunflower and the solar cells are presently more reliable.

Other work in progress at LRC includes:

Phillips' work on a magnetic stabilization system for vehicles within the influence of the earth's magnetic field. A small bar magnet is supported in a series of gimbals. When alined suitably relative to the earth's magnetic field, the interactions of the two fields can produce or stabilize an orbital vehicle. A working experimental apparatus has been constructed and investigation is proceeding. It is said to be accurate to several seconds of arc.

LRC is investigating the problem of condensation at zero gravity using a gas jet in the center of a moving liquid annulus.

It was noted that Instrument Research Division (IRD) is working on Environmental Control System (ECS) requirements for a 60-day mission in connection with Applied Materials Physics Division (AMPD) space station investigations.

AMPD is planning some experimental investigations on the stabilization and heat balance problems associated with an inflatable, rotating space vehicle configuration which provides a small artificial gravity.

The <u>Lewis Research Center</u> work related to Apollo is as follows:

Lewis Research Center will be investigating the Marquardt PAT-C system for attitude control. They had originally planned to do their preliminary checkout work in an altitude tunnel ($\frac{1}{2}$ in. H_2O), but have now decided that they should immediately start working in a hard vacuum. LeRC has a 4 ft × 5 ft "moon-dust" chamber in which investigations are being run on the jet dust kickup expected on lunar landings. It is planned to connect this to a 30-inch diameter diffusion pump and instrument the chamber for the PAT-C tests. Extensive instrumentation is needed for thrust measurement during the short pulses. A major system problem is to insure simultaneous arrival of propellant and oxidant at the chamber. The propellant and oxidant valves are not presently linked mechanically. Tank pressures will be initially 300 psia; later, Marquardt wants to use several thousand psia. These high pressures may present a problem. It may be that pumps weigh less.

Lewis Research Center has developed a number of "small" bipropellant systems of 50 to 100 pounds thrust.

Lewis Research Center is studying a lunar landing vehicle. Initial tests are proposed to be conducted by drops from aircraft at Wallops Island. Drop altitude would be about 40,000 feet. The vehicle is spherical in shape, has a retrothrust rocket for velocity control. Attitude control is accomplished by a cold gas jet system.

Marshall Space Flight Center Apollo-related activities:

Attitude and velocity control-reaction nozzles.- Small variable thrust engines can be used to control pitch, yaw, and roll and the velocity vector of a vehicle. MSFC has subcontracted the study and development work to U.S. Naval Ordnance Test Station, at China Lake, California, NOTS.

Variable thrust, zero to full thrust, and pulse thrust mode operations are studied. Storable propellants, red fuming nitric acid and unsymmetrical dimethyl hydrazine, are used.

A variable area injector controls the flow of both fuel and oxidizer simultaneously and in proper proportion to obtain full range of thrust control. Only one moving part, a cylindrically-shaped pintle with conical faces controls the propellant flow into the combustion chamber.

Hypergolic propellant combinations are used eliminating the need for a separate ignition system.

By means of suitable hydraulic or electrical actuation, the pintle can be moved to full open, partially open or closed position of the valve.

Larger engines can be controlled by a hydraulic system, small engines may use solenoid action, high torque D-C-motor and screw, very small engines - magnetostriction linear actuators.

Test - Experience at NOTS.- Zero to 1,350 pounds thrust engines using hydraulic actuation have been operated successfully at frequencies as high as 20 cps.

The hydraulic servomethod becomes somewhat impractical if such motors are reduced in size.

Some work has been done at NOTS on direct electrical actuation:

a. Solenoid action (fig. 3 in report NOTS, Nov. 9, 1960) to open and close the pintle of a small thrust engine to operate it in a pulsed mode at moderate frequencies.

Solenoid action does not appear at this time to be practical due to the high currents required for this actuation.

b. The possibility of using direct motor drive and screw has been investigated (see fig. 4 in report). The method appears to be somewhat cumbersome though not as power consuming. Reaction times are also slow unless motors of very high torque to weight ratio become available. Miniaturization is necessary.

c. A third method of pintle actuation has recently been demonstrated in a laboratory device. It is based on magnetostrictive elements. If a bar of some magnetostrictive metal such as nickel is placed in a magnetic field, the bar will constrict along its length. This constriction may amount to several millionths of an inch per inch. Since a single constriction is not sufficient to actuate the injector pintle, a method of compounding this motion through repetitive constriction has been developed. Applying two magnetic clamping devices, one at each end of the construction bar, and by proper phasing of constriction and clamping at a high cycle rate, a linear motion will then be produced in the bar. Forces of sufficient magnitude to actuate the injector mechanism may be produced.

A first working model of a linear magnetostrictive actuator has been produced.

Some improvements on the clamping devices to yield high forces are presently incorporated. It is slippage in the clamping mechanism which now prevents the present actuator from achieving the forces which it should be capable of developing. (See reports: Western Radiation Laboratory, LA7, date Mar. 15, 1960, Aug. 8, 1960.)

Objectives of work at NOTS. - Proceed on Research and Development of small variable thrust engines with the following objectives:

a. Demonstration of a pulsed or digitally-operated variable thrust engine of approximately 20 pounds maximum thrust using a magnetostrictive linear actuator for control.

b. Demonstration of a small variable thrust engine which will have a propellant leakage rate no greater than 1cc/day under a vacuum of 10^{-3} mm Hg.

c. Development and demonstration of a combustion chamber design for use with variable thrust engines capable of being fired for at least 120 seconds, with no more than 10-percent throat area increase and of a size and weight commensurate with the thrust level of the engine. The engines have graphite nozzles.

d. The demonstration of one or more methods of achieving propellant ratio compensation in small motors.

References are:

1. Rutkowski, E. V.: An Investigation of Small Demand Thrust Motors for Space Vehicle Application. Status Report of the Naval Ordnance Test Station, China Lake, California.

2. Anon: Report dated Mar. 15, 1960 from Western Radiation Laboratory, Los Angeles, Calif. to NOTS on the use of magnetostriction for actuation of small thrust motor injector valves.

3. Anon: Report dated Aug. 8, 1960 from Western Radiation Laboratory, Los Angeles, Calif. to NOTS on the use of magnetostriction for actuation of small thrust motor injector valves.

Auxiliary power.- MSFC has broad experience in turbomachinery, pumps of different kinds, hydraulic systems and components, propellant feed systems, etc., and can offer support in practical solutions for specific problems connected with nonpropulsive power for space vehicles.

General information on the state of the art of power conversion systems is gathered, no special research is pursued in this field.

Mechanical elements in space.- Solar energy collectors, antennas, radar screens, etc., require mechanical elements exposed to the outer space environment and to be manipulated from the inside of the pressurized vehicle.

Design studies on hermetically-sealed mechanical linear and torque-motion transmitters are carried out presently. The cases of actuating a valve and driving a hermetically-sealed pump shaft located in the space vacuum are being considered in particular.

The friction problem in space mechanisms is a major one.

The function of all mechanisms on earth basically depends upon friction; in some cases friction is desired, in others it is undesirable. The latter is the case for all bearings. By means of suitable material combination (lubricants between sliding surfaces) the friction may be reduced considerably.

In most cases, the average designer is not aware of the fact that the principle of lubrication is not only a matter of material properties, but to a great extent based on the existence of the atmosphere.

Since there is no atmosphere in space, new design principles have to be applied for bearings. Research has to be conducted to find these new design principles.

Studies made so far by MSFC in this field have analyzed the mechanism of friction and lead to conclusions for the development of space mechanisms.

Further R and D programs are needed to replace the forces which cause seizing of surfaces in intimate contact by friction-eliminating forces which are available in space.

MSFC also submitted to the panel the following discussion of a possible recovery system for SATURN which is under investigation at MSFC.

Flexible wing glider.- Recovery of the first and possibly the second stages in the SATURN program will be of great value to any space program. Recovery of the S-1 stage has been in effect since the start of the SATURN program.

In the beginning of the SATURN program, the recovery system employed a parachute system that provided for the recovery of the booster from the ocean. This system was preferred because it made use of components and/or techniques that were within the state of the art, and consequently it would reduce development costs and yield higher reliability. Also, vehicle modifications to accommodate the recovery system were held to a minimum.

During the course of the recovery system development, funding problems were encountered and the recovery system development program was postponed. The postponement gave an opportunity to investigate other recovery concepts. A number of unsolicited proposals were received by the Recovery Project Office, and among these were two similar techniques utilizing the Rogallo flexible wing concept. The two proposals were submitted by the Ryan Aeronautical Company and the Los Angeles Division of North American Aviation, Incorporated.

MSFC is presently drafting a statement of work to conduct studies on the paraglider. The purpose of this study is to investigate the technical and economical feasibility of using a paraglider (Rogallo flexible wing) in connection with the recovery of the SATURN booster and upper stages. Main emphasis of the study will be directed toward

[redacted]

the S-I stage application and, after the feasibility of this application has been established, effort should then be directed to the recovery of the S-II stage (the second stage of the SATURN C-2 configuration). Second-stage recovery is of interest, since, for orbital missions, a two-stage fully recoverable vehicle is extremely attractive economically.

The scope of this study will require detailed preliminary design of a complete paraglider recovery system, detailed drawings of the method of attachment to the SATURN S-I stage, a complete operational and cost analysis of the system, and a detailed plan for the required research and development of such a system. The results of this study will be detailed enough to permit selection of the most promising recovery system for the SATURN C-2 at the earliest possible date. A study period of 6 months is envisioned for this effort.

The proposed recovery system consists of a flexible deployable wing based on research by Mr. Francis Rogallo and others at NASA, Langley Field, Virginia. The flexible wing was chosen because of the advantages of mission launch volume, long glide range, simplicity, lightweight, and controllability.

Unlike the conventional wing composed of a rigid skin covering a forming structure, the flexible wing is composed of a membrane of flexible material attached to three supporting members. The center keel and the two leading edges, which can be of rigid structure or inflatable material, are joined at the foremost point to define a triangular envelope. The edges of the flexible membrane are attached to the side members, and the membrane is joined to the keel throughout its length at the centerline. The flexible wing is joined to the vehicle by means of cables or rigid structural members.

The most obvious advantage of the flexible wing is its extremely low storage volume and lightweight. If inflatable keel and side members are employed, exceptionally high storage densities may be obtained. The lightweight construction allows more effective wing area per unit weight than a conventional wing resulting in greater lift for a given wing weight. Test and analysis show that the wing possesses a high inherent stability about all three axes.

Several methods of flight control are possible. The center-of-gravity shift (fore and aft for pitch, to the side for roll) is simplest The desired center-of-gravity shift can be obtained simply by reefing or payin, payout, of supporting cables or struts. Control can also be affected through camber changes obtained by adjusting the nose, flexing of the wing material, application of canard surfaces, and application of surface control through membrane doors, spoiler plates or conventional rudder and elevators.

[redacted]

Preliminary studies indicate that the weight of the complete paraglider system will be around 7,000 or 8,000 pounds, depending on whether an inflatable wing or a three-wing tandem-connected design with rigid structural members is employed, respectively.

Research and Development needed for Apollo includes work in the following areas:

Rotor recovery systems -

a. Although conventional helicopters are not very stable, and somewhat difficult to fly in a power-off condition, a rotor recovery system could probably be designed for the specific case which would be adequate.

b. The use of suitable instrumentation would help the situation greatly.

c. If cyclic control is to be used, the vehicle must be capable of azimuth control and stabilization.

d. Systems using rotor tip power were discussed.

e. A system was suggested which incorporates autorotative descent to 20 to 50-foot altitude, and solid propellant rotor tip rockets for the touchdown phase.

f. The work of Kaman (USAF-sponsored), Wright Air Development Division, Vertol and Bell should be closely monitored.

The Environmental Control System requirements need considerably more detailed definition.

Work in Reaction Controls by LeRC and MSFC is good.

Inertia wheels and magnetic stabilization systems should be investigated. LRC is working in these areas.

The ships system power supply questions need further resolution. For Apollo, the shielding penalty of nuclear systems appears unacceptable. The question of the use of solar systems or chemical systems (either fuel cells or expansion engines) remains open. It was noted that the integration of chemical systems becomes complex because of the possibility of tie-in to abort power, the use of fuel for shielding, etc. It was suggested that NASA should study fuel cells in more detail for this application.

LRC has been working on studies and testing of power systems suitable for Apollo.

One area in which Research and Development is needed is materials for equipment in space, particularly organic polymers for use in inflatable antennas, solar reflectors, electrical and thermal insulation, sealants, seals, parachutes, pyrotechnics, as examples. There are many important questions about stability, strength, and compatibility which are unanswered. LeRC is doing some work in the field. LRC is also active in the field. MSFC has a contract with the National Research Corporation, Cambridge, Mass., to develop sealants.

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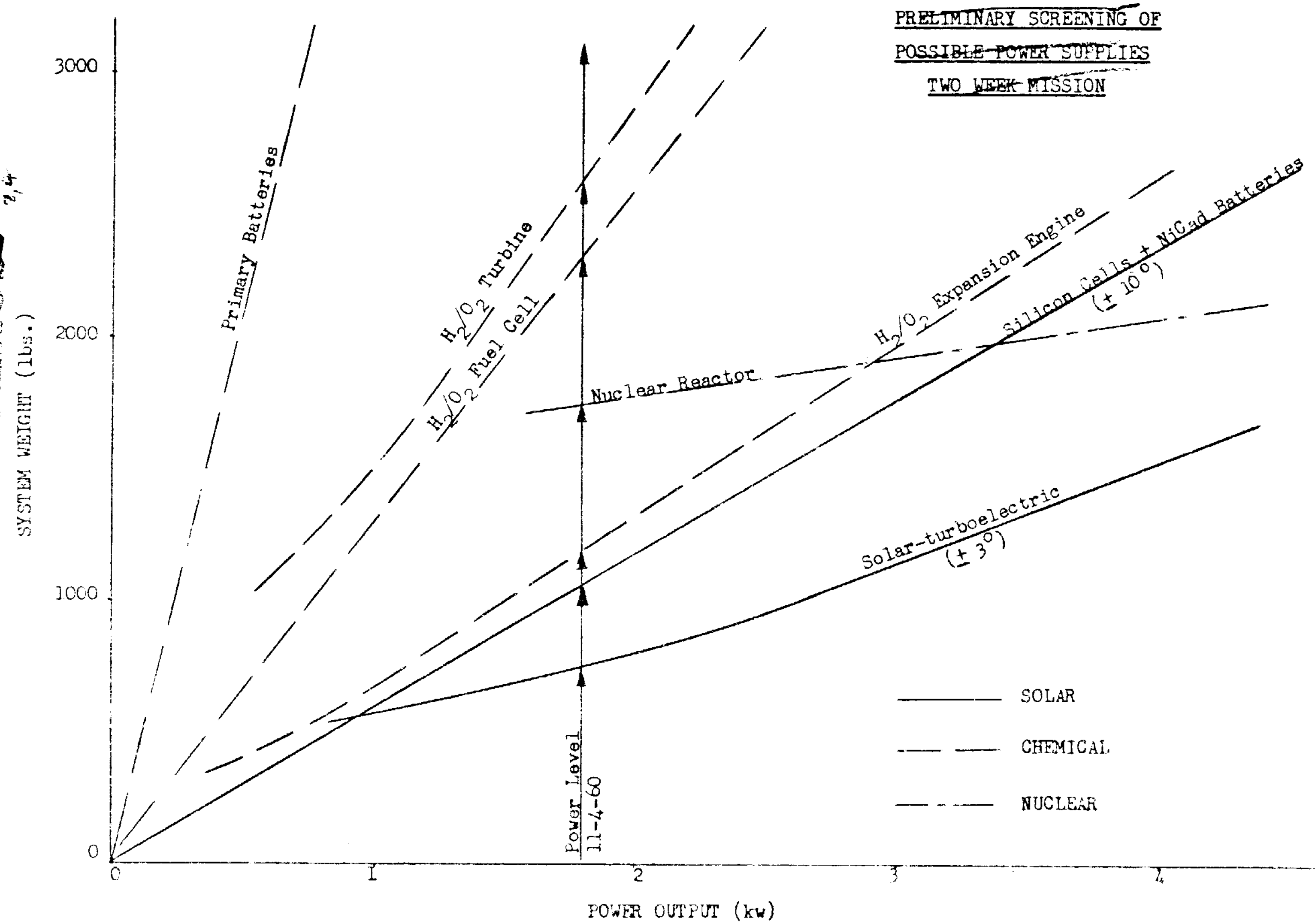

Section III, Figure 1.- Preliminary screening of possible power supplies 2-week mission.

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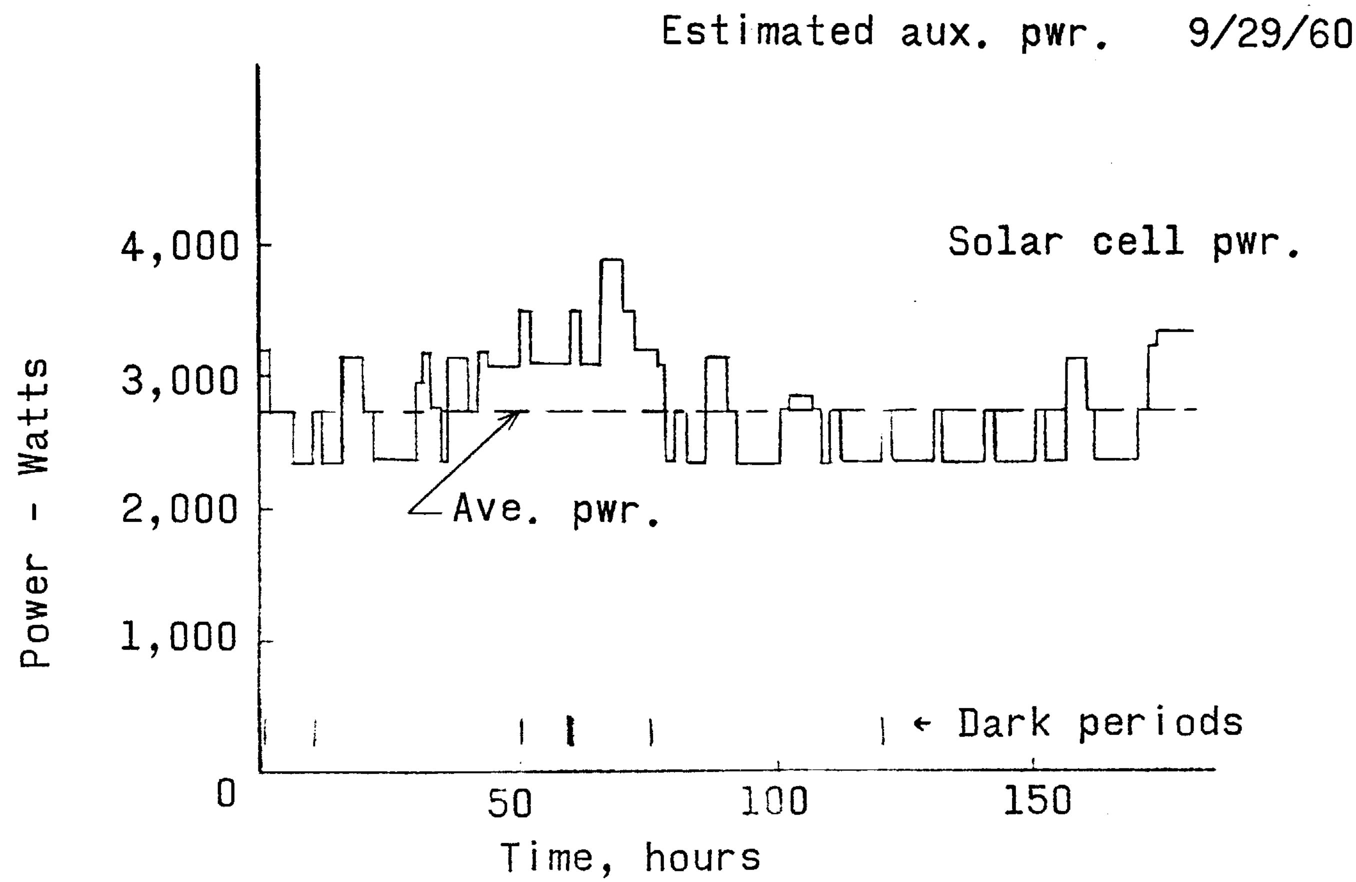


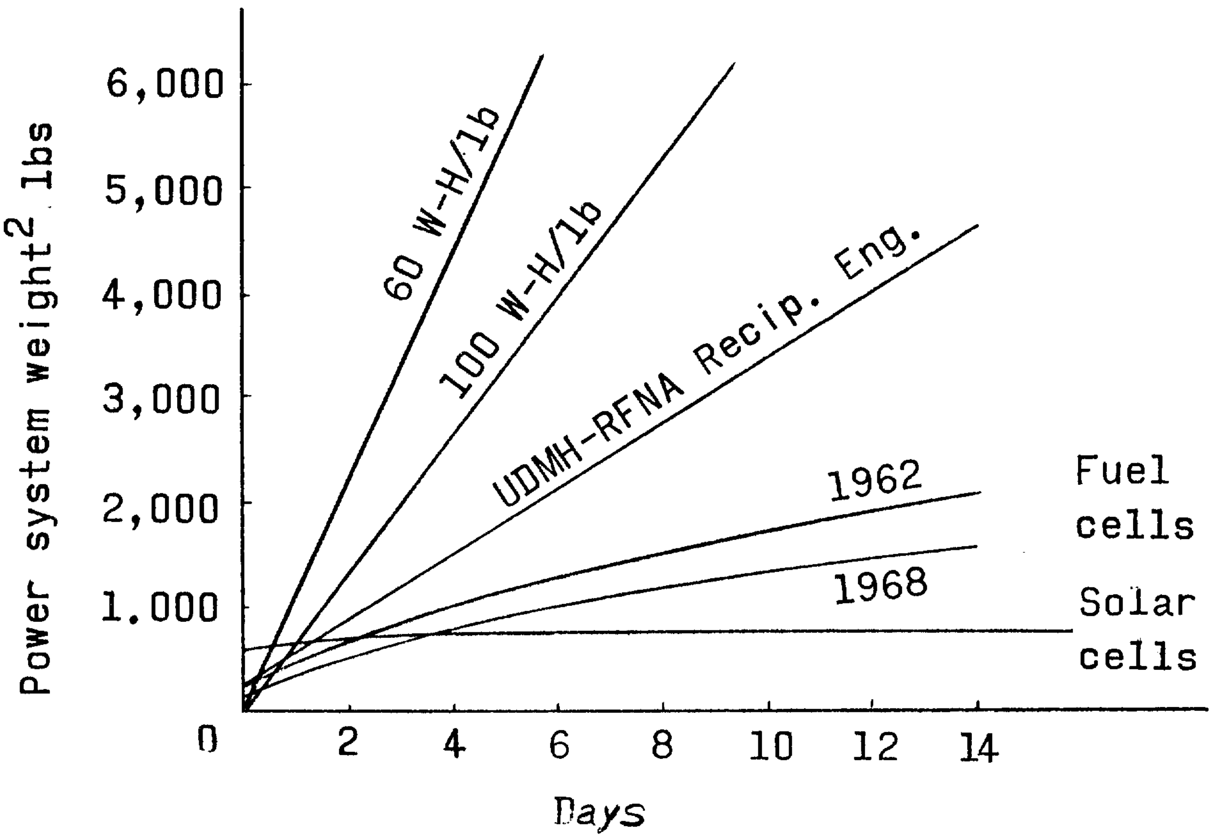
Section III, Figure 2.- LRC Discussion presented at Apollo Mechanical Systems Technical Liaison Group Meeting No. 1 - 1/6/61.

Weights for a Space power system

Avg. pwr. 2.75 KW

Peak pwr. 4.00 KW

PRIMARY BATTERIES



Section III, Figure 3.- LRC Discussion presented at Apollo Mechanical Systems Technical Liaison Group Meeting No. 1 - 1/6/61.

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SECTION IV

EXCERPTS FROM

PROJECT APOLLO MINUTES OF MEETING

OF

TECHNICAL LIAISON GROUP - HEATING

January 11, 1961

Ames Research Center

Moffett Field, California

[redacted]

a. Marshall Space Flight Center (MSFC).- Mr. Dahm discussed some exploratory studies of heat protection systems for vehicles returning from lunar flight. Lifting trajectories with a lift-drag ratio of 0.5 were used, with reentry angles from the overshoot boundary corresponding to a maximum of 10g. Distances from the reentry point (taken at an altitude of 100km) ranged from about 1,500km to about 12,000km and times from about 280 seconds to about 2,000 seconds. A blunt cone (22° with a length-diameter ratio of 0.5) and a hemisphere were assumed as representative vehicle shapes with ballistic parameter W/C_DA from 500 to 50 lbs/ft^2.

Inner surface-cooled quartz, with vaporizing water as coolant, was found to be a promising method for heat protection. A shield having a moderate weight with flexible characteristics is provided. The weight of a shield made of solid quartz, including coolant, is about one-third to one-half the weight for a Teflon shield. A further significant weight reduction can be achieved by use of porous quartz. For a blunt cone having a solid-quartz shield of optimum thickness distribution, the combined shield and coolant weight was about 20 to 25 percent of the total weight of the reentry body. For the hemisphere, this weight was about 5 to 6 percent of the total weight. The heat shield required for the overshoot boundary was also found suitable for much steeper trajectories with decelerations beyond human tolerance. At the stagnation point, approximately 50 percent of the heat was absorbed by radiation and about 50 percent by mass transfer. Only a small fraction was absorbed by coolant. Practically all of the ablated material was vaporized. It should be noted that radiation was not included in these calculations.

Work is continued to extend the investigations with exact solutions on shields of varying surface density of quartz at stagnation points as well as other points on the body. Solutions are also sought for large nose radii with sizable shock-layer radiation. Mr. Connell of MSFC discussed work related to ballistic reentry nose cones having a W/C_DA of 1450. Fused amorphous silicon materials with variable porosity were used. Studies have also been conducted of composite materials employing both high- and low-temperature ablators with plastic or foam honeycomb insulation material being used.

b. Langley Research Center (LRC).- An analytical study has been made using three types of reentry trajectory maneuvers. The types considered for parabolic reentry velocities are constant C_2 maneuvers (with and without roll after pulling up to horizontal attitude) and a constant g maneuver after reaching g_{max}. When plotted in terms of:

$$\frac{\dot{q}\sqrt{R}}{\sqrt{W/C_D A}} \quad \text{against} \quad \frac{Q\sqrt{R}}{\sqrt{W/C_D A}}$$ these

results show a related family of hyperbolas.

Three of the projects mentioned were planned free-flight tests at velocities of approximately 29,000 fps using Scout booster vehicles. One test will determine the heating rate of a blunt body by the thin-walled technique using an Inconel skin. Upon failure of the Inconel skin, measurements of ablation rate of a secondary surface of Teflon will be made. The second flight test will evaluate the integrity of the external char of deep-charring ablators. The general external shape of the configuration is the same as that mentioned for the first test. The third test will obtain total and spectral distribution of radiation. This test is still in the preliminary stages and the instrumentation has not been completely defined.

c. Ames Research Center (ARC).- Three fundamental problem areas of reentry heating for lunar vehicles were summarized:

(1) Forebody heating - The heat-transfer rates and heating loads to the forward-facing surfaces of a lunar vehicle are understood reasonably well if the air in the shock layer and the boundary layer are in thermodynamic and chemical equilibrium. Theoretical methods which have been developed are quite sound if the flow is laminar, and these methods have been verified in shock tubes at least up to enthalpies near those corresponding to satellite entry. Experimental verification at speeds above satellite is, however, lacking. For the case of turbulent flow, the theoretical approach is very weak and little or no experimental results are available in the speed range of interest. It is expected, however, in the flight regime where the aerodynamic heating rates are most intense that the flow over the forward surface of a reentering lunar vehicle will be laminar as the Reynolds numbers rarely exceed 2 or 3 million.

The effect of thermodynamic and chemical nonequilibrium on the heating rates has been assessed in reference (1). In this investigation the flight regimes where the flow at the boundary layer edge, or within the boundary layer will be equilibrium, have been estimated. The conclusion is that inviscid flow at the boundary-layer edge near the stagnation point will be equilibrium for most of the flight cases of interest, but that the flow in the boundary layer in general wil not be in equilibrium. The effect of this boundary-layer nonequilibrium on the heating rate for a noncatalytic surface has also been assessed in this work. The main effect of nonequilibrium flow was found to be one of relieving the severity of the problem, the extent of the relief depending upon aerodynamic parameters of the vehicle, and the maximum deceleration encountered during the flight. This work considered only surfaces of zero catalytic effectiveness.

Surfaces having finite catalytic efficiency have been studied (reference (2)). This study estimates the effect of surface catalytic activity upon the heating rate. The main result of the study is that even materials having relatively high chemical activity will not allow recombination of all the atoms which reach the surface, as the process, particularly at high altitude, is limited by the diffusion rate of the atoms across the air boundary layer. It is also shown that glassy materials such as quartz will have little or no chemical activity at the wall. The analyses outlined in references (1) and (2) consider in addition to the stagnation point on the body the effect of expanding the flow over the forward-facing surfaces and give methods whereby these effects can be readily calculated. It should be emphasized that this work is theoretical and has not been experimentally verified except in a rather crude way.

In addition to the previously-mentioned investigations of the effects of nonequilibrium in the boundary layer, one other investigation is currently underway which includes the effect of chemical and thermodynamic nonequilibrium effects in the shock layer. In this analysis, the gas in the shock layer is assumed to be heat-conducting, viscid, and chemically reactive. The conservation equations are integrated between the body surface and the shock wave. The objective of this investigation is to determine the boundaries in terms of flight altitude and flight velocity between equilibrium and nonequilibrium flow in the shock layer and to investigate the effects of nonequilibrium chemistry and thermodynamics on the standoff distance and its implications in terms of the convective and radiative heating. Only preliminary results are available at this time. These preliminary results do indicate, however, that the standoff distance is a function of the body surface temperature and the body diameter. This is in contrast to the result obtained from the inviscid analysis where the ratio of the standoff distance to the body radius is essentially dependent upon the enthalpy and the Mach number only. The reason for the differences between the real gas case and the equilibrium inviscid gas case is that when the gas is heat-conducting and chemically reacting, the effect of the wall temperature can be felt in the shock layer at distances much beyond the normal boundary-layer thickness. The effect on the convective or radiative heat-transfer rates have not at present been evaluated. In summary, it appears that the heat-transfer rates to forward-facing surfaces, if these surfaces are relatively smooth and do not contain abrupt corners, can be predicted reasonably well.

If the forward-facing surface does contain corners or regions where the flow expands very rapidly, theory is unable, at the present time, to account for these effects. Two investigations have been completed in which these effects have been measured experimentally. One investigation reported in reference (3) compared the heat-transfer results on the M-1 configuration obtained in a Mach No. 6 low-temperature wind tunnel with those obtained in a shock tunnel where the flow was at a relatively high temperature. The main result of this analysis seems to indicate that the distribution of heat transfer around a complicated body shape can be determined from cold wind-tunnel tests, but that the level of the data will be altered by the enthalpy of the airstream. The second investigation is one where the heat-transfer distribution around flat-faced cone was measured. It was noted that the theoretical predictions of heat transfer vary markedly from the data in the region where the flow is rapidly expanding around a corner, and that the measured heat-transfer rates do not reflect a total magnitude predicted by the theory. This difference may be due in part to the experimental difficulties or may be due to the fact that the boundary layer may separate around these corners, in which case these effects are not included in the theory. This investigation is presently in rough-draft form and is listed in reference (4).

(2) <u>Radiative heat transfer to stagnation region of blunt bodies</u> - At the speeds characteristic of lunar return vehicles, the air in the shock layer is heated sufficiently to cause thermal radiation from the hot gas to strike the stagnation region of a body in relatively large amounts. Rough indications of the magnitude of the heating were presented in reference (5) and in some cases, the radiant heating rates can be of the same order as the convective heating rates. These theoretical predictions were mainly based on the data of AVCO, reference (6). To extend the AVCO data to high speeds and over a wider range of densities, the radiation from the gas cap of small models fired at high speeds in the supersonic free-flight range have been measured. The thermal radiation presented in this figure has been normalized as indicated by the theory of Kivel, outlined in reference (6). Radiation power is plotted against flight velocity and it is interesting to note that the general theoretical predicted trends are borne out by the experiments. A relatively large degree of uncertainty does exist in the experimental measurements of these quantities, and it is emphasized that these results are very new and unpublished. Of particular interest is a group of measurements obtained in the vicinity of 22,000 feet per second, which lie above the general level of the majority of the data and the theory. These data were obtained under low-pressure conditions where it was suspected that the radiation resulted from an out-of-equilibrium situation in the gas cap. The pertinent point here is that these

values may be 30 to 40 times those which result from an equilibrium radiation.

The distribution of radiation heat transfer around shaped bodies is in the process of being calculated by the ARC Fluid Mechanics Branch, using these radiation data. Radiation heating along the centerline of a delta wing at angle of attack has been calculated. These calculations indicate that the radiation intensity is a maximum at approximately 90 percent of the wind cord. Again it is emphasized that these calculations are preliminary and are continuing.

In addition to calculation of radiant intensity incident on a body, calculations of the shielding effectiveness of oblating on absorbing gases from the surface of the body have been made. These results have been reported in references (7) and (8), and indicate that for cases where radiation heating is important, protection may be obtained by suitable choice of ablation material. In this connection, an apparatus to experimentally investigate ablation materials which may be used as protection against combined radiation and convective heating is being built at ARC. This apparatus consists of a combination arc-wind tunnel and an arc-image furnace. The model is placed in the test section of an arc-heated wind tunnel which supplies the convective heating. Radiation from an arc-image furnace is also focused upon the front face of the model. The radiation intensity can be varied independently of the convective heating rate. The convective heating rate, which this apparatus can generate, is well within the range encountered by lunar vehicles and radiation rates, which can be focused upon the body surface, can also be varied over sufficient range to cover most flight conditions. It is hoped that this apparatus will be in operation in the near future in order to obtain some experimental verification of the theories given in references (7) and (8), and in addition to investigate the properties of various ablators as they are subjected to radiation fluxes.

(3) <u>Heat transfer to afterbodies emersed in a separated flow</u> - Three investigations are currently being carried on at ARC of the general problem of heat transfer in bodies emersed in a separated flow. One investigation being conducted in the Fluid Mechanics Branch will measure experimentally the local heat transfer in the wake of a two-dimensional body. The models are arranged such that various base configurations can be installed. These consist of a flat base, a reentrant base, and a wedge-shaped base. The models have been completed and the tests will be run shortly.

The second investigation is one which will measure experimentally the base heating and the base pressure distribution on modified Mercury capsules. This investigation will be conducted in the ARC 12-inch helium tunnel, at Mach numbers of from 8 to 26, and over a range of angles of attack. The models are completed and the tests will be run as soon as calibration runs are completed at the ARC 12-inch helium tunnel.

A third investigation, theoretical in nature, is underway which is aimed at explaining and predicting distribution and magnitude of the heat transfer in a separated region. The approach used in this investigation follows the method of Chapman, reference (9), to account for the external boundary layer and uses an integral method to account for the boundary-layer growth along the solid body at the bottom of the separated region. The essentially inviscid flow in the separated region is treated as a vortex. A balance of momentum, energy, and continuity around the entire separated region is postulated. Integral methods are used to evaluate the local distribution. Preliminary results indicate that the distribution of heat transfer in a separated region is given reasonably well by this analysis. However, the level of the data is not predicted very well. The theory is being altered to account more realistically for the actual flow field within the separated region, and results from this second attempt should be available shortly.

The following references were used during the Ames summarization:

1. Goodwin, Glen, and Chung, Paul M.: Effect of Nonequilibrium Flows on Aerodynamic Heating During Entry Into the Earth's Atmosphere From Parabolic Orbits. Preprints, Second International Congress for Aero. Sci., Zurich, Switzerland, Sept. 1960.

2. Chung, Paul M., and Anderson, Aemer D.: Heat Transfer to Surfaces of Finite Catalytic Activity in Frozen Dissociated Hypersonic Flow. NASA TN D-350, 1961.

3. Reller, John O., and Seegmiller, H. Lee: Convective Heat Transfer to a Blunt Lifting Body. NASA TM X-378, 1960.

4. Terry, James E.: Convective Heat Transfer to a Lifting Flat-Faced Cone Entry Body. NASA TN D-775, 1960.

5. Yoshikawa, Kenneth K., Wick, Bradford H., and Howe, John T.: Radiation Heat Transfer at Parabolic Entry Velocity. Joint Conference on Lifting Manned Hypervelocity and Reentry Vehicles. Apr. 1960.

6. Kivel, B., and Baily, K.: Tables of Audiation from High Temperature Air, Res. Rep. No. 21. AVCO Res. Lab., 1957.

7. Howe, John T.: Radiation Shielding of the Stagnation Region by Transpiration of an Opaque Gas. NASA TN D-329, 1959.

8. Howe, John T.: Shielding of Partially Reflecting Stagnation Surfaces Against Radiation by Transpiration of an Absorbing Gas. Prospective NASA TR.

9. Chapman, Dean R.: A Theoretical Analysis of Heat Transfer in Regions of Separated Flow. NACA TN 3792, 1956.

Further study is believed needed in the following areas:

(1) One area in need of additional attention for experimental investigation is the heating on control surfaces. Other significant areas are being pursued as rapidly as existing techniques allow.

(2) Determine the relative importances of the unknowns in the heating area by relating estimated "ignorance" factors to resulting weight penalties in the spacecraft.

(3) Evaluation of radiant heat inputs and their effects upon the ablation heat shields.

(a) Calculation of magnitude of radiation loads using most pessimistic assumption for lunar mission and assess importance in terms of heat-shield weight (assuming no non-equilibrium effects).

(b) Tests of ablators subject to these combined loads to determine ablation rates to check current theories.

(4) Heating on control surfaces.

(a) Methods to predict control heating in shear flows.

(b) Wind-tunnel tests at high Mach numbers to obtain methods (even crude ones) of prediction.

[redacted]

SECTION V

EXCERPTS FROM

PROJECT APOLLO MINUTES OF MEETING

OF

TECHNICAL LIAISON GROUP - GUIDANCE AND CONTROL

January 12, 1961

Ames Research Center

Moffett Field, California

[redacted]

Ames Research Center Efforts in Guidance and Control
Applicable to Apollo

Preliminary studies have been conducted to determine the problem of ranging from a partially illuminated disk. These studies indicated that by use of concentric rings in the field of view the center of the illuminated disk could be measured manually to within 3 seconds of arc. The results of studies of altitude disturbances caused by passenger movement indicate that crew movements provide very little net disturbing torque.

Broader coverage of many pertinent research programs appears in the Ames Report on Research of Interest to the NASA Committee on Control, Guidance, and Navigation, dated September 27 to 28, 1960 by Mr. Howard F. Mathews. Another reference is Ames Research Projects in Support of the Manned Lunar Mission, dated May 1960 by Mr. Alvin Seiff.

a. Pilot's effectiveness in controlling a reentering lunar vehicle.- Analog simulator study with pilot in the loop to control lift and bank to achieve point landing from visual display of vehicle maneuver capability footprint. Onboard computer uses Chapman equations and assumes measured or computed values of present position, velocity and acceleration. Pilots have achieved good controllability and outlined areas of critical control sensitivity particularly near overshoot boundary. This is a continuous prediction technique and a fixed or reference trajectory technique has been set up and is now being run for comparison.
STATUS: Report in editorial. Will be extended to skip maneuver and higher L/D.

b. Longitudinal range extension by skip reentry maneuver.- Analytical and digital computer study of reentry in two dimensions when skip to 400 miles (safely below radiation belt) is permitted. Complete dynamic equations: $L/D = \pm 0.5$ and $L/D = \pm 2$ cases completed. Nonskip range of 2,000 to 5,000 miles extended from 2,000 to 16,000 miles for undershoot boundary entry, and up to slightly over one orbit at overshoot boundary. Exit angle and velocity are quite critical.
STATUS: Two dimensional writeup available. Extension to three dimensions and rotating earth contemplated.

c. Effect of lateral and longitudinal range capability on the reentry window for the lunar mission.- Analytical study of allowable variations of reentry time and orbit inclination permissible for point landing as a result of ±500 miles lateral and from 2,000 to 10,000 miles longitudinal range capability.
STATUS: Initial results available as curves.

d. Reentry corridor modification by continuous variation of lift and drag.- Digital computer study. Desirable deceleration

[redacted]

investigated. Two-dimensional study so far with an extension of Chapman's analysis. Heating effects considered. Trade-offs for increased corridor width.
STATUS: Report in preparation. Investigation continuing.

e. Abort velocity requirements for the lunar mission.- Analytical study of abort velocity requirements for the Saturn lunar boost trajectory. Severe fuel penalty when burnout occurs at higher altitude or velocity, or larger flight-path angle.
STATUS: Report in progress. No further work planned.

f. Inertial guidance stability and accuracy for midcourse and reentry from the lunar mission.- Digital computer investigation of inertial system with onboard or earth-based corrections from 60,000km to touchdown.
STATUS: Preliminary analytical work.

g. Pilot's ability to control in zero and high g environments.- Experimental investigation of human pilot ability to perform meaningful control tasks in high g centrifuge environment and in low and zero g flight tests.
STATUS: Johnsville centrifuge tests in April. Zero g tests in F-104B Flight Test Center in March.

h. Manned air-bearing capsules.- Experimental investigation of attitude control of manned capsules in low-friction air-bearing supports. Control, stabilization and instrumentation system research.
STATUS: Limited three-degree-of-freedom iron cross operational soon. Nine-foot sphere under construction.

i. Thrust requirements for time of arrival and orbit inclination variation with constant perigee.- Use of Kepler's equation for impulsive thrust trajectories. Also, treatment of L/D variation to modify lunar reentry trajectories.
STATUS: Curves available. Will be incorporated in ARS summary paper by Eggers.

j. Midcourse guidance using onboard measurements with smoothing.- Analytical study of methods of determination of present state from onboard measurements, prediction of future states, formulation of guidance laws. Extension of optimal linear filter theory. Measurements contaminated with statistical noise.
STATUS: Digital computer investigation being programed.

[redacted]

k. Onboard pilot observation and computation, emergency midcourse navigation technique.- Conic trajectory determination by pilot from photographic sighting and hand computation a possible emergency mode.
STATUS: Photographic reduction and hand computation achievable in reasonable time. Accuracy being investigated.

l. Midcourse trajectory measurements and error analysis.- Conic trajectory determination by various measurements with instrumentation errors.
STATUS: Closed form geometric data available. Time measurement iteration schemes being investigated.

m. Space vehicle attitude control studies.- Analytical, analog, and experimental investigation of inertia wheels, integrating gyros, current coils, micro jets, and twin gyros for attitude control of satellites - space fixed Orbiting Astronomical Observatory - earth pointing Nimbus.
STATUS: Inertial wheel report issued NASA TN D-691. Magnetic coil report in preparation. Gyro report in preparation. Twin gyro preliminary analytical work. Micro jets - preliminary experimental work.

n. Manual onboard optical tracking.- Experimental investigation of accuracy in measuring angle to center of a disk or crescent from a reference direction by using a pilot-operated theodolite mounted on a stable platform, and in turn supported in a large gimbal structure to simulate spacecraft motions.
STATUS: Preliminary measurements from a fixed base indicate probable accuracies of about 3 seconds of arc.

o. Ground-based optical tracking with T.V. system.- Experimental investigation of possibility of optical tracking of a lunar vehicle using a 12-inch telescope and T.V. system employing an image orthocon.
STATUS: Equipment due by March.

p. Automatic star-tracking equipment.- An analytical and experimental investigation of principles and techniques for automatic onboard star tracking for space vehicle guidance and attitude control, using various types of primary sensors, scan methods and gimbal arrangements.
STATUS: Preliminary work in progress.

q. Spacecraft attitude disturbances due to passenger movement.- An analytical study of space vehicle motion resulting from passenger movements within vehicle.
STATUS: Preliminary curves available.

r. Lunar trajectories and point return.- An analytical and digital computer study of direct ascent circumlunar trajectories with return to a specified landing area. Practical constraints on time and direction of Saturn launch from AMR considered in detail. Guidance sensitivity and fuel requirements studied.
STATUS: Writeup available.

s. Lunar mission nonretrograde trajectories.- A simplified analytical study of lunar orbiting trajectories that are not retrograde. Overall mission using salvo launch of two Saturns for rendezvous and a lunar landing is explored.
STATUS: Report ready for editorial.

t. Rendezvous terminal guidance.- Analytical study of a proportional navigation homing technique for terminal guidance. Based on position and velocity data.
STATUS: Report ready for editorial.

u. Rendezvous thrust requirements.- Analytical study of impulsive thrust for rendezvous.
STATUS: Completed.

Additional Apollo-related work in progress at ARC includes:

Simulation studies to develop a concept for a terminal guidance system including displays, controls and an autopilot. One of the objectives is to determine how well a human subject can control the reentry vehicle.

Since the flight regime is restricted to within the atmosphere, the relatively simple Chapman equations can be used for simulation and prediction. The inputs to the system are calculated accelerations. A null display of the "footprint" on the earth type is used. The operator's task is to keep his desired landing spot as near as possible to the center of his maneuverability "footprint." Accuracy is within about 5 percent of the excursion from the null.

Reference was made to the following recent Ames reports considered to be of interest in the Apollo studies:

1. Creer, B. Y., Smedal, H. A., Wingrove, R. C.: Centrifuge Study of Pilot Tolerance to Acceleration and the Effects of Acceleration on Pilot Performance. NASA TN D-337, Nov. 1960.

2. Smedal, H. A., Creer, B. Y., Wingrove, R. C.: Physiological Effects of Acceleration Observed During a Centrifuge Study of Pilot Performance. NASA TN D-345, Dec. 1960.

[redacted]

3. Smedal, H. A., Holden, G. R., Smith, J. R., Jr.: A Flight Evaluation of an Airborne Physiological Instrumentation System, Including Preliminary Results Under Conditions and Varying Accelerations. NASA TN D-351, Dec. 1960.

Langley Research Center (LRC) work in Guidance, Control and Rendezvous:

a. Aero-Physics Division -

(1) Trajectory Analysis of Reentry Vehicles with Realistic Drag Polars (based on tunnel tests of L-series bodies) at escape velocity. Includes multiple pass and parking orbits. Also lifting Mercury. In progress.

(2) Numerical calculations of the atmospheric portion of lunar return trajectories with and without lift modulation.

(3) Calculations of accessible landing areas for lunar return vehicles of various modes of operation.

b. Applied Materials and Physics Division -

(1) Attitude Control Systems for Space Stations and Lunar Mission Module.

c. Aero-Space Mechanics Division - Flight Mechanics Branch-

(1) Midcourse Guidance for Rendezvous.

(2) Launch Conditions and Trajectories for Rendezvous and Return. Being published as NASA TR.

(3) Abort Procedures During Saturn Launch. Covers choice and consequences of various methods of applying abort fuel in $1 \leq \bar{u} \leq \sqrt{2}$. Report now in editorial stages.

(4) Evaluation of Several Methods of Interplanetary Navigation. A comparison will be made on the basis of accuracy and equipment required. The methods thus far considered differ chiefly in the nature of the celestial bodies observed. One method requires the observation of two stars, the sun and one planet, and a second requires the observation of the sun and three planets. Both of these methods require measurement of time as well as angle between bodies. A third method is being considered which does not require time to be measured.

[redacted]

d. Guidance and Control Branch -

(1) Investigation of a Three-Degree-of-Freedom Attitude Control System for a Satellite Vehicle. System analyzed in NASA TN D-626 is mechanized on an air-bearing platform. Presently, servocontrol of magnet is being extended to allow control about any axis.

(2) Study of a Wobble Damper for a Rotating Space Station. Analog computer studies are being made, and an inertial simulator is being built under contract to study damping of oscillations of a rotating space station. Simulator to be constructed by July 1961. Gyroscopic precession is used to supply the control torques.

(3) Lateral and Longitudinal Range Control for a Vehicle Entering the Atmosphere of a Rotating Earth. Analytical studies completed of entry from orbital velocities using altitude control to a reference trajectory and lateral control as a function of heading error. Accurate control possible to about 85 percent of range capabilities of vehicle.

(4) Reentry Range Control Using Terminal Controller Techniques. Analytical study completed of range control using a linearized prediction technique. Report is in preparation. This method appears promising and will be extended to include lateral control, and entry from parabolic velocity.

(5) Guidance of a Space Vehicle Approaching a Planet along an Entrance Corridor. Study of NASA TN D-191 is being extended to include use of dead band in guidance logic to reduce corrective velocity required. Results indicate that dead band reduces accuracy with little, if any, saving in fuel. Results to be discussed in paper for AAS in Dallas, Texas, January 17, 1961. Further plans include study of effect of instrument accuracies.

(6) Determination of Conditions for an Attempted Soft Lunar Landing and Return to an Orbiting Vehicle. Analytical studies of penalties associated with simplified techniques for lunar soft landing and return to orbiting vehicle. Reports in preparation.

(7) Analog Simulation of a Pilot-Controlled Rendezvous. Analog simulation of terminal phase (last 50 miles) with six-degree-of-freedom. Report in preparation.

(8) Automatic Control of the Terminal Phase of a Rendezvous. Analog computer studies of two-control techniques for terminal phase (last 50 miles). Control technique uses correction to place ferry vehicle on colliston course, then a braking phase using continuous or intermittent thrust. Report in preparation.

(9) Study of Time Required to Detect Motion of Objects Moving on a Contrasting Background. Use of optical techniques to detect angular rate of line-of-sight is being studied with application to pilot control of a rendezvous.

(10) Analytical Investigation of an Adaptive Control. Analytical and analog study of an adaptive control which adjusts system gain to maintain constant response to a sinusoidal test signal. Application to aircraft and launch vehicle control studied. Report in preparation. Method appears highly satisfactory.

(11) Investigation of Stability and Deployment Characteristics of a Drag Brake for the Mercury Capsule. Analytical study in progress of a device similar to a kite tail intended to reduce time in orbit in case of failure of retrorockets.

e. <u>Theoretical Mechanics Division</u> -

(1) Stability and Control of Lunar Reentry Vehicle Configurations. A study of dynamic stability characteristics (lateral, longitudinal, and six-degree-of-freedom) and aerodynamic control effectiveness parameters is being made for earth-entry conditions. Of four LRC-proposed configurations, considerable results have been obtained to date for the L-1 vehicle.

(2) Manned Control of Lunar Vehicles During Earth Entry. A man-controlled analog-simulator investigation is to be made of control and guidance problems associated with earth entry for several possible Apollo configurations from a lunar orbit. Currently work is proceeding on the analog program and instrument panel.

f. <u>Instrument Research Division</u> -

(1) A Study of Tracking Problems, Equipment, and Accuracy Requirements.

[redacted]

(2) Experimental Study of Optical Measurements. Will use a telescope having a diaphragm and a split image to measure the sun's disk, the moon's disk, and two close stars. Will investigate limits on diffraction of optical systems at different ends of the light spectrum. Also alinement accuracy based on (known) resolution.

Flight Research Center (FRC).-

a. Current programs.-

(1) A study of factors affecting the pilot's capability during exit and reentry conditions of varying severity is planned utilizing the performance capability of the X-15 airplane. Flights during the expansion of the airplane envelope will provide the opportunity to obtain data for this investigation. Later tests specifically for this investigation will be made to more extreme exit and entry conditions. About 5 minutes of zero g can be obtained with the large engine version of X-15.

(2) A program to investigate the reaction control requirements for space vehicles is being conducted using F-104 and X-15 airplanes. Reaction control systems being investigated are: on-off, proportional, reaction-control augmentation, and rate command. During the program with the F-104 airplane, dynamic pressures of the order of 6 lb/sq ft have been obtained. The X-15 program will begin in the spring of 1961. The reaction control systems used will have a maximum thrust of 90 lbs. Maximum altitude reached for the F-104 tests was 87,000 ft.

(3) Supersonic and hypersonic handling qualities are being studied using the X-15 as the research vehicle. Also, a comparison of theoretical, wind tunnel, and flight stability and control derivatives will be made over the flight-envelope capability of the airplane. Comparisons up to a Mach number of 3.0 have been made and show good agreement.

(4) The effect of extreme altitude and zero g environments on systems is being studied using the X-15 and F-104B airplane. Tests, to date, have used the F-104B airplane. A number of equipment problems have developed under these conditions.

(5) An adaptive control system is being developed by Minneapolis-Honeywell Regulator Co. for the X-15 airplane. Several modes, i.e., fully adaptive, rate command, normal

[redacted]

acceleration command, and altitude hold, will be available for evaluation in flight by the pilot. The system has already been evaluated on the X-15 six-degree-of-freedom flight simulator. With this system the handling qualities of the airplane were greatly improved. Flight tests are scheduled to begin in September 1961.

(6) A continuing program of FRC concerns the assessment of simulation requirements in various flight regimes. Pilot opinions of airplane handling in various regimes, including the extreme semiballistic trajectories of the X-15, will be correlated with ground simulation in an attempt to better define simulator requirements.

(7) A broad study is being sponsored by WADD to define the energy-management system requirements for various space vehicles. If feasible, the system resulting from this study will be tested in the X-15 airplane. Also, the determination of terminal-guidance requirements and the landing characteristics of low lift-drag ratio configurations is being studied at the FRC.

(8) Measurements of physiological information on pilots during flights to extreme flight environments are being made in an attempt to correlate pilot performance, opinion, and workload, and also to determine the effects of extreme flight environments on the pilot. Both X-15 and F-104 airplanes will be used to provide the desired environment.

(9) Flight evaluation of a ball or hot nose as an air data source is being conducted. Comparison with more conventional sensors is in progress.

(10) A variable stability airplane (F-100C) in which the pilot can change the stability and damping over wide limits is being used to study the control problems associated with the X-15, Dyna-Soar, and other vehicles.

(11) A program has been initiated to investigate the control requirements of the Dyna-Soar vehicle using an analog simulation of control conditions to be expected during the Dyna-Soar trajectory. Preliminary results indicate that the basic Dyna-Soar vehicle control is marginal over much of the angle of attack (0° to 50°) and Mach number (0.2 to 20) range capability of the vehicle. This is attributed, primarily, to very light damping and severely coupled motions resulting from aerodynamic control coupling, dihedral effect, and

coupling due to angle of attack. Simple rate dampers made the vehicles controllable in most regions, but adverse control moments and coupling made precise maneuvering difficult. However, an adaptive control system with rudder interconnect provided near-optimum handling characteristics over the operating envelope of the vehicle.

(12) A study of the display requirements for the accurate stabilization of a piloted vehicle being used as a platform for astronomical observations has been initiated. Quickening is being studied as a means of increasing the efficiency of the pilot Manual Control System.

b. Completed programs.-

(1) Simulator Investigation of Orbital Rendezvous Problems - NASA TN D-511.

(2) Utilization of a Pilot in the Launch and Injection of a Multistage Orbital Vehicle. Reports published: IAS Preprint No. 60-16 and U.S. Air Force NASA Conference on Lifting Manned Hypervelocity and Reentry Vehicles - Part II, Paper 20.

(3) Review of Techniques Applicable to the Recovery of Lifting Hypervelocity Vehicles. U.S. Air Force NASA Conference on Lifting Manned Hypervelocity and Reentry Vehicles - Part I, Paper 20.

(4) Simulator Studies of Jet Reaction Controls for Use at High Altitudes - RMH58G1a.

(5) Flight Controllability Limits and Related Human Transfer Functions as Determined from Simulator and Flight Tests. Proposed NASA TN.

(6) An Analytical Approach to the Design of An Automatic Discontinuous Control System. Proposed NASA TN.

(7) Controllability of the X-15 Research Airplane with Interim Engines During High-altitude Flights. Proposed NASA TN.

MSFC Apollo-related Guidance and Control efforts:

a. Attitude Control Systems.-

A series of feasibility studies have been made towards the stabilization of vehicles in space environment. The systems studied are based on stabilization by inertial flywheels plus expulsion devices. Originally the studies were based on the 24-hour satellite application which had unique problems of its own, but essentially the results of the studies are applicable to Apollo. For the 24-hour satellite, power and weight were of utmost importance plus the fact they were to operate in vacuum. Theoretical analysis, simulations, and later, the design and construction of instrumentation to be mounted and tested on the satellite motion simulator was accomplished. At present a single-axis system is working on the simulator and a three-axis system should be operating in February 1961. From these studies should develop complete design specification for attitude control of vehicles of three different weights.

Two developments in instrumentation are at present going on. These are:

(1) Transistorized commutator motor (brushless) which will operate in vacuum.

(2) Magnetically-suspended spherical flywheel. A ring-shaped model is working at present in the laboratory.

In addition, a number of engineering programs are going on in the field of attitude sensing such as horizon stellar sensors for attitude determination and navigation. Contract with Barnes Engineering Company has been initiated to develop a universal horizon seeker which gives attitude sensing and altitude. Another is a feasibility study and contractual supervision of a lateral velocity over altitude meter (V/h).

b. Inertial Equipment.-

A four-gimbaled inertial platform is being developed and a prototype model will be built this year and should be tested and flown as passenger on SA-III or SA-IV.

c. Guidance Theory.-

Most of the work done in this field is that due to Dr. Hoelker and his group in the Aeroballistics Division. The system as being developed there is to be universal in

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that it will adapt itself to wide variations in flight conditions such as may arise in a multiengine and multistage rocket system with fuel minimized. The development involves an enormous computation program and has not been worked out in full detail. The pertinent equations are in series form and from this arises the instrumentation problem - how many terms of the series are necessary for the required accuracy? This determines the size of the onboard computer. The system is being extended and applied to midcourse navigation and reentry into the earth's atmosphere. The system should be more completely defined in 3 to 6 months. The engineers in Guidance and Control Division have made a preliminary study on instrumenting the system, and indications are that the computer requirements will not be unreasonable.

For application to midcourse, Dr. Schultz-Arenstoff is adapting the same methods for the complete orbit from injection to reentry. This will require about 6 months.

A separate study is being made of the reentry phase both in Guidance and Control Division and in the Aeroballistics Division. These studies are based on a path-control technique, and involves a rather simple system compared to the adaptive types. First results indicate an accuracy of ±15 kilometers.

A block diagram for the Saturn guidance system was presented which also included the generalized guidance equations. (See fig. 1.)

The reference of the following reports was noted:

(1) Kennel, Hans F., and Drawe, G. P.: Feasibility Study of an Attitude Control System For Space Vehicles. Rep. No. DG-TM-17-59, Army Ballistic Missile Agency, (Redstone Arsenal, Ala.), Apr. 6, 1959.

(2) de Fries, P. J.: Horizon Sensor Performance in Measuring Altitude Above the Moon. NASA George C. Marshall Space Flight Center, Jul. 15, 1960.

(3) Hartbaum, Dr. Helmut K.: One-Axis Radio Attitude Sensor. Reps. Nos. TWP-M-G and C-I 60, NASA George C. Marshall Space Flight Center, Sept. 8, 1960.

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(4) de Fries, P. J.: Analysis of Error Progression in Terminal Guidance for Lunar Landing. NASA George C. Marshall Space Flight Center, Sept. 15, 1960.

(5) Webster, John L., and Schultz, David N.: Study of a Simplified Attitude Control System for a 24-Hour Satellite. Reps. Nos. M-NN-M-G and C-7-60, NASA George C. Marshall Space Flight Center, Sept. 30, 1960.

(6) de Fries, P. J.: Guidance Concept for Lunar Landing, NASA George C. Marshall Space Flight Center, Oct. 10, 1960.

(7) Hoelker, R. F.: Theory of Artificial Stabilization of Missiles and Space Vehicles With Exposition of Four Control Principles. Rep. No. MTP-M-AERO-60-7, NASA George C. Marshall Space Flight Center, Nov. 7, 1960.

(8) Anon: Terminal Phase of Soft Lunar Landing (Data Sheets). NASA George C. Marshall Space Flight Center, Nov. 1960.

(9) Webster, John L., and Schultz, David N.: A Three-Axis Study of a Flywheel Type Attitude Control System Progress Report. NASA George C. Marshall Space Flight Center. Jun. 30, 1960.

(10) Webster, John L., and Schultz, David N.: Progress Report Space Vehicle Attitude Control Systems Analysis. NASA George C. Marshall Space Flight Center. Feb. 15, 1960.

Jet Propulsion Laboratory (JPL) noted that the JPL Space Progress Summaries contain a full exposition of JPL's Apollo-related Guidance and Control efforts and then proceeded to discuss the Guidance and Control for Ranger, Surveyor, Mariner and the Mars Probe.

The Attitude and Control System is essentially the same for all three vehicles. It consists of cold gas, N_2 jets at 15 psia. The solenoid valves are operated in an on-off fashion. Control is by means of rate gyros and sun and earth sensors. N_2 is stored at 3,000 psia and is dropped to the nozzle pressure of 15 psia through a single-stage pressure reducer. Maximum angular accelerations are low, on the order of 2×10^{-4} radius/sec^2.

The Sun and Earth Sensors will be arranged about the spacecraft in such a way that there is only one attitude which will produce a simultaneous null in all sensing arrays. "Clairex" photoconductor cells will be used which have been precision matched for temperature characteristics. Nortronics is producing primary sensing assemblies (designed by JPL) which have an angular accuracy of $\pm 0.05^{\circ}$. These sun sensors consist of a matched pair of photoconductor cells, each of which is partially shaded by one end of a "shadow bar" which is raised above the surface of the spacecraft. At null, the photocells are equally illuminated. One assembly is used for roll, one for yaw. Earth-sensing is accomplished by an array of three photomultiplier tubes arranged in a triangular pattern and each partially shaded by a "T"-shaped shadow bar consisting of a staff and a crossbar parallel to, and slightly raised from, the spacecraft surface. Two of the photomultipliers are located on the spacecraft surface below the intersections of the edges of the staff and the crossbar of the "T". The third photomultiplier is located on the surface of the spacecraft at a point equidistant from the other two and such that it is half shaded by the side of the crossbar opposite from the staff. The photomultiplier outputs are chopped, demodulated, then sums and differences are combined to provide control signals for roll and for earth-pointing antenna alinement. One interesting feature of this system is that it will function even when the earth is not completely illuminated. This is due in part to the fact that the antenna beam width includes the entire earth disk at any appreciable distance.

Other secondary sensors will be located about the spacecraft to provide course control for earth, sun or star acquisition.

The same system will be used for both Mars and Venus except that for Mars a star reference will be used since the earth is poorly lighted as seen from an earth-Mars trajectory. While near earth, a star close to the ecliptic pole will be used and near Mars, Mars will be used. The star-Mars reference would be used for roll control and a stored program would be used for control of the earth-pointing antenna.

Midcourse corrections are initiated by radio command from earth. The spacecraft will perform a pair of single-axis, gyro-supervised turns, then thrusting will commence. The midcourse engine will have a thrust of 50 lbs. Control is by means of four vanes in the jet servoed to the gyros so as to null angular accelerations. After thrusting, the system would automatically revert to the reacquisition and cruise mode.

The following studies are being conducted by STG in support of the Apollo Guidance and Control requirements. These studies are in an early phase at the present time.

a. <u>Predictor-type Guidance Scheme for Abort and Reentry Navigation</u>.- Consists of a prediction computer solving simplified equations of motion of reentry and display to the man a "footprint or fan." This scheme is being investigated for use in controlling skip.

b. <u>Apollo Abort Separation Study</u>.- This study is intended to determine the effect of lift and engine gimbaling arrangements on abort thrust level and impulse requirements.

c. <u>Two-body Prediction Midcourse Guidance Study</u>.- Determination of system accuracy and fuel consumption.

d. <u>Preliminary Lunar Landing Study</u>.- Determination of guidance system to effect a lunar landing.

e. <u>Position Computer Study</u>.- A mechanization of a system to provide position and velocity information.

A brief discussion of the objectives of each of these programs was given. The intended result is to obtain a complete guidance scheme that can be evaluated for error propagation.

A number of suggestions for Research and Development required for Apollo were discussed, including:

a. An "absolute emergency" navigation system in which the crew would use only a land camera and a sliderule.

b. Can radio ranging be used to reduce the accuracy requirements for celestial observations? Would such a composite system fall within the limits set by the Apollo guidelines?

c. It was noted that studies had been performed on the effects of crew motions within the spacecraft on attitude alinement and control requirements. The effect was found to be quite small. Such a study might also be conducted for rotating machinery onboard.

d. Problems of planet tracking when the planetary disk is only partially illuminated must be studied further.

e. A study should be made of the transient effects of guidance updating by external information.

f. The effects of artificial g configurations on observation and guidance should be investigated.

g. It was suggested that a "how goes it" or mission progress evaluation display should be developed for the crew suitable for an entire mission.

h. An abort guidance scheme is needed including an abort decision computer and pilot display. It was noted that William Roger Teague has been working on an "abort sequence" scheme.

i. It was suggested that earth-orbit evaluation of the position computer input be accomplished in a highly eccentric orbit (500 to 1,000-mile perigee; 60,000-mile apogee).

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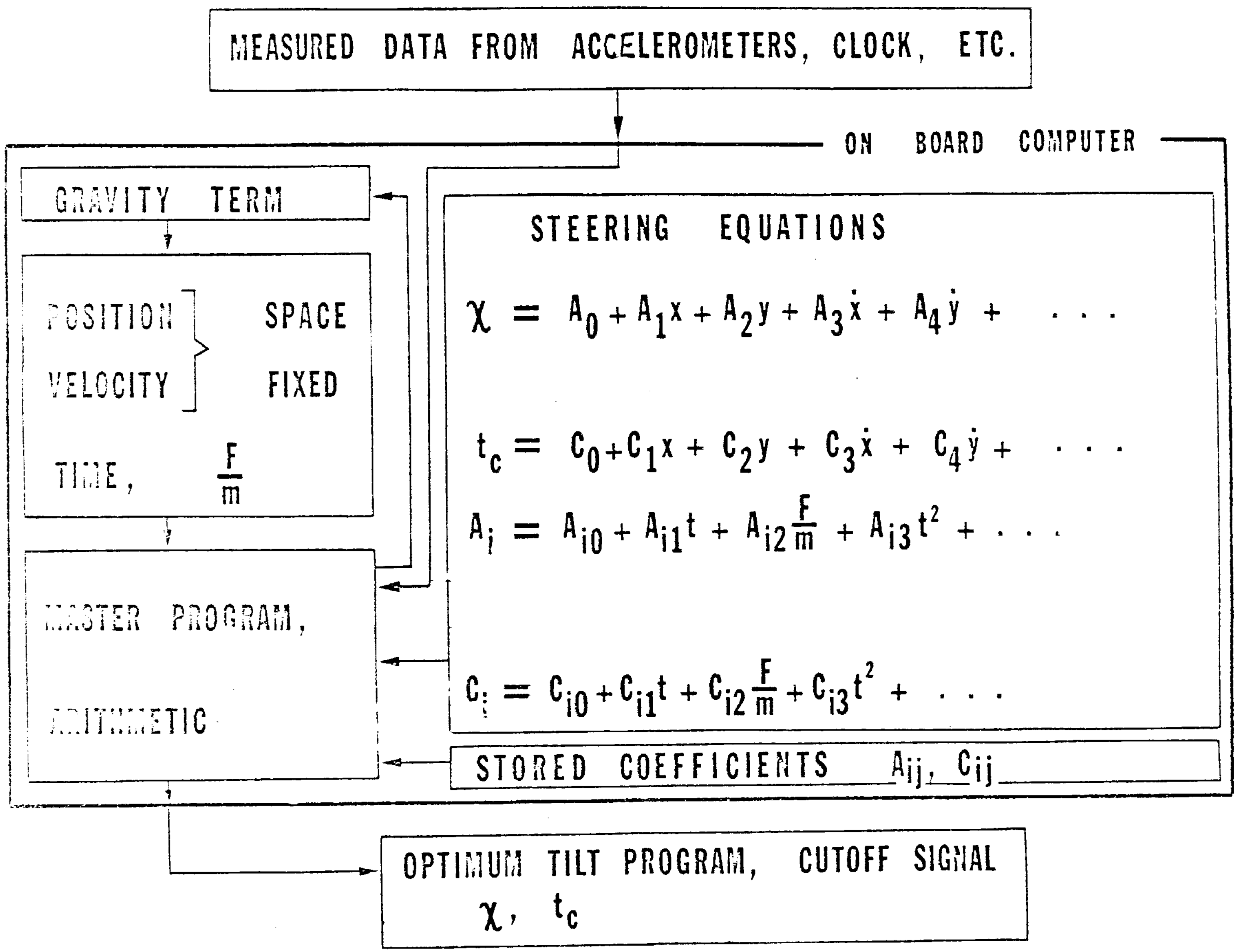

GUIDANCE SYSTEM SCHEMATIC

Section V, Figure 1.- Guidance system schematic.

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SECTION VI

EXCERPTS FROM

PROJECT APOLLO MINUTES OF MEETING

OF

TECHNICAL LIAISON GROUP - CONFIGURATIONS AND AERODYNAMICS

January 12, 1961

Ames Research Center

Moffett Field, California

[redacted]

a. Space Task Group (STG).- A short motion picture was shown of the hard surface and water landings of a lenticular model. Horizontal landings were made directly on the curved heat shield. The model behavior during hard surface landings indicate landings of this nature to be feasible but violent motions resulted upon water impact. Most of the group felt this reentry vehicle presented a promising landing concept and warranted further investigation.

Apollo Working Paper No. 1006, entitled, "An Analysis of the Errors in Position Given by an Onboard Lunar Navigation System Using Observations of Celestial Bodies" was referenced.

b. Ames Research Center (ARC).- Projects pertinent to the Apollo mission. The configurations investigated to date include:

(1) Mercury-type capsule

(2) Ames M-1

(3) Ames M-2

(4) Ames flat-faced cone

(5) Disk-type (lenticular)

(6) Dyna-Soar and other winged configurations

In addition, some aerodynamic tests of boosters considered to be of interest were included.

The Ames M-1 has received more extensive investigation of the configurations, and most of the results are presented in an in-house report. It was noted that little emphasis at Ames has been directed towards investigations of parachute landing systems.

c. Marshall Space Flight Center (MSFC).- The primary effort concerning the Apollo vehicle is the aerodynamic influences of various Apollo shapes on the booster during ascent. A preliminary study has been made of a sphere-frustum configuration with control flaps. Also discussed were the aerodynamic characteristics of spherically blunted cones. Aerodynamic design charts have been prepared for spherically blunted cones. These data were taken from both published and unpublished wind-tunnel data, and, where possible, were compared to existing theories. Hypersonic values as predicted by the Newtonian theory were also included.

d. Jet Propulsion Laboratory (JPL).- The status of work at JPL pertinent to configuration and aerodynamic group activity was outlined. JPL is engaged in the development and operation of unmanned lunar, planetary and deep space probes. The aerodynamic research and development at JPL is concentrated in the following subjects:

(1) Entry into planetary atmospheres from both hyperbolic and elliptic orbits

(a) Mars hyperbolic entry

(b) Venus hyperbolic entry

(c) Return to earth of lunar material samples

(2) Aerodynamic constraints on spacecraft from launch to injection

(3) Advanced gas dynamic research

e. Langley Research Center (LRC).- Work now in progress or proposed at Langley pertinent to Apollo is divided into the following general categories: trajectories and rendezvous; guidance and control; instrumentation and data transmission; propulsion; auxiliary power; configurations, reentry aerodynamics, and heating; structures and materials; dynamic loads; environmental hazards.

Langley configuration studies were briefly discussed.

(1) The Langley configuration studies (for the reentry vehicle) began with four basic shapes designated L-1, L-2, L-3, and L-4 and were selected so as to encompass widely varying shapes with the hope of exposing any major advantage or disadvantage of a particular approach. All vehicles were designed to have hypersonic $(L/D)_{max}$ of the order of $\frac{1}{2}$, internal volumes of the order of 350 ft^3, and maneuver, trim, and control capability from the angle of attack for $(L/D)_{max}$ (lowest α limit) to angles of attack up to and in some cases beyond 90° angle of attack. The current versions of these vehicles are sketched in figure 1; all have undergone changes and fixes as the result of problem areas exposed in earlier tests, and all the versions shown here have been tested hypersonically.

The L-1D vehicle is an outgrowth of an extensive parametric study of both round-bottom and flat-bottom half cones begun about 1 year ago, in which such features as face cant, cone half angle, etc., were studied. To give an example of problem areas that have been encountered and remedied by fixes in the course of tests of the L-series vehicles, figure 2 shows how sweeping the flap-hinge line on the L-1 type eliminated most of the adverse yaw generated by roll control.

The L-2 vehicle was included in the overall program because there was some early feeling based on preliminary estimates that this approach might prove to be the lightest vehicle, and secondly, this approach tended to take advantage of any Mercury technology that appeared suitable for the Apollo mission. Parametric studies of face and corner radii and flap sizing have been made and are continuing on this approach. It should be mentioned here that the letter designations A, B, C, etc., shown in figure 1 affixed to a vehicle designation, do not imply that only a certain number of modifications have been explored; for example, while L-2C is the currently selected version of the L-2 type, versions L-2 through L-2E have been tested (the original version of any type had no letter suffix).

The L-3 approach was included since a delta-shape reentry or lifting face was found in preliminary heating studies to give the lowest total heat load; further, the heating and aerodynamic characteristics have received considerable study. The implication here was that this approach might prove to be one of the lightest.

The L-4 approach was built primarily around a catering toward what was felt to be the most efficient stacking and packing of equipment, etc. In this connection, layouts of the equipment were made with sufficient accuracy for all vehicle types to insure that the internal volume and vehicle size were reasonable, and to determine the region within which the center of gravity might be reasonably positioned.

(2) Vehicles L-5 and L-6 were slenderized L-1 types, having $(L/D)_{max}$ values of the order of 1 and $1\frac{1}{2}$; both were eliminated from further study early in the program. Two vehicles have been added to the program recently. These are a STG proposal of the lenticular type which LRC has designated L-7, and the blunted full cone proposed by the MSFC which LRC has designated L-8. Both L-7 and L-8 have undergone some tests at LRC; these vehicles are shown in figure 2.

(3) Vehicles L-1 through L-4 are envisioned as reentering at high angle of attack with a nominal L/D of the order of 0.3 or so and by varying α and L/D and maneuvering, would make

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the necessary corrections to reach the desired landing point. The landing system proposed for these vehicles is a paraglider final descent (beginning at high subsonic speeds) with a parachute backup system. The paraglider would make a flare landing with the order of a foot or so per second vertical velocity and about 30 to 40 ft/sec horizontal velocity (no wind). In the event that landing as made without flaring (emergency condition), maximum vertical velocity would be about 16 ft/sec and maximum horizontal velocity about 60 ft/sec (with 30 ft/sec wind). The L-8 vehicle envisions much the same landing system as the L-1 through L-4 systems, but reenters at small angle of attack, and its pitch and yaw maneuvers are limited to about $\pm 11^{\circ}$.

The L-7 vehicle is envisioned as entering at fixed attitude ($L/D \approx 0.3$) and with no employment of aerodynamic controls during reentry. When low speeds are reached (high subsonic or low supersonic) rigid surfaces are unfolded from behind the rear portion of the vehicle that would give it conventional landing capability at a prepared landing strip; it would belly-in to a skidding landing at about 160 ft/sec horizontal velocity (no wind) and a foot or so per second vertical velocity.

(4) Reports are now in preparation on the aerodynamic characteristics of these vehicles, and it is beyond the scope of this summary to attempt even a summation of this work. The current versions of the L-1 through L-4 vehicles have generally good static longitudinal and directional characteristics and trim capability over the required α range. The main difficulties have centered on control and cross-coupling. Vehicles L-3A and L-4A are still saddled with problems of this nature, and while one could keep chipping away and making fixes, it seems unwise if these vehicles do not offer some special advantages over the others. Furthermore, it appears that the objective of freezing the approach by late March 1961 is still sound. Accordingly, LRC has dropped the L-3 and L-4 approach from further study.

(5) In comparing weights of reentry vehicles, it has been assumed that each will have the same equipment and same internal systems weight. With this assumption, weight comparisons can be made on the basis of only structure plus heat protection weights. Such comparisons have been made with inputs from the Structures and Aero-Physics Divisions at LRC, based on a common trajectory. Further refinements are currently being made to the heat load calculations along with studies of the influence of different trajectories. Currently, the picture for the L-1D, L-2C, L-7, and L-8 is about as follows for the weight of the structure plus heat protection plus basic landing system. (No backup system has

[redacted]

been included since this is assumed to be a chute of the same weight for all. The basic system for the L-7 is the extendable rigid surfaces, and for the other vehicles it is the paraglider.)

Vehicle	Weight, lbs (Structure + heat protection + basic landing system)
L-1D	3,000
L-2C	3,000
L-7	3,300
L-8	3,800

These weights must be regarded as tentative, but they should be indicative within ±200 pounds. On this basis there is little debatable difference between the first three vehicles. The L-8 will undoubtedly be heavier because of its high heat protection weight that is associated with its W/C_DA being some 4 or 5 times that of the other vehicles. (Recall that it is proposed to operate at $\alpha \approx \pm 11^{\circ}$.) If advanced heat protection systems come along with unit weights sufficiently small to cause significant reductions in the L-8 heat protection weight, or should these few hundred pounds additional weight not be a cause for concern to the booster capability, the L-8 appears to be a highly competitive approach and LRC feels it should remain in the competition. Of course, whatever weight benefits can be counted on for L-8 from reduction in heat protection unit weights by use of new ablation materials would also apply to the other vehicles; thus, the heat protection weight should always be less for the other vehicles. The total weight of these reentry vehicles would probably range between 6,000 and 7,000 pounds.

(6) With regard to the L-7 vehicle, after discussions at LRC, it was concluded that the fixed attitude approach is not desirable. However, this could probably be taken care of in some manner by reaction and aerodynamic controls similar to those that would be used on the L-2C. The major objection to the L-7 appears to be its landing mode and the associated hazards. It must reach its preselected and prepared landing site on land, or some other prepared site on land, to make its 160 ft/sec belly landing a success; at least this is the indication of tests to date and the consensus of opinion at LRC. Water landings appear most hazardous. Furthermore, except for being tailored to this belly-in landing feature, L-7 is basically little different from the L-2C approach when the capability for attitude variation is added to L-7. The feeling at LRC is that if the paraglider shows the same type of reliability in large-scale

tests (a large-scale program is getting under way at LRC in connection with both Saturn recovery and final phase letdown and landing for reentry vehicles) that it has achieved in small-scale tests, the potential advantages of this system outweigh other systems. No insurmountable problems have been encountered in preliminary engineering designs of large para-gliders. What is needed now is reliability demonstrations with unmanned and manned drops. This is planned.

Primarily because of the possible landing hazards associated with L-7, it has also been dropped from further consideration by LRC.

(7) The vehicles that are currently under consideration at LRC are shown in figure 3. Additional studies are in progress on these vehicles. The dynamic stability characteristics are of particular concern, and this area is being examined both analytically and experimentally, including such studies as the effect of the edge radius on L-2C and the ability of the extended flaps to damp. Although other liaison groups have the responsibility in these areas, it should be mentioned that extensive analytical and experimental heating studies together with trajectory studies are continuing.

Of the three vehicles shown in figure 3, two have axisymmetry while one, L-1D, does not. There are some advantages to a symmetrical vehicle in connection with mating the vehicle to the module, loads during launch, etc. There are also some advantages to asymmetry of the type exhibited by L-1D such as during an abort soon after launch where negative lift could be achieved passively, rather than through vehicle rotation by reaction jets, to give the desired clearance between the vehicle and the following booster so as to avoid collision or close proximity to booster explosion.

Note also that L-1D and L-8 have positive lift-curve slopes whereas L-2C has a negative, lift-curve slope. This may reflect on the outcome of the dynamic stability studies now in progress.

It should have been mentioned earlier that all the vehicles under study have leaned toward matching, or nearly so, the 10-foot upper stage diameter of Saturn C-1 and <u>four-stage</u> Saturn C-2. L-1D is about 10 feet in length by $12\frac{1}{4}$ feet in width; L-2C is 11.6 feet in diameter; L-8 is $10\frac{1}{2}$ feet in length and $10\frac{1}{2}$ feet in diameter. All the vehicles are envisioned as being mated to the top of the mission module. The access hatch for L-2C would be constructed through the heat shield. This offers no complication to speak of, as has been confirmed by the design of such hatches at LRC and by several industrial groups. The storm cellar walls contain guidance and in-flight abort fuel to serve the additional purpose of shielding from the radiation produced by low energy, high flux, solar events.

f. Flight Research Center (FRC).- Activities at FRC related to Project Apollo. Most of the activities at FRC are concerned with the X-15 and Dyna-Soar configurations. It was noted that initial Dyna-Soar configurations made use of escape capsules, but the vehicle is now the primary escape mode. Land landings are the primary operational mode and water landings are treated as an emergency condition (ditching).

It was pointed out that analytical studies of the landing characteristics of a lenticular vehicle indicate that the pilot should have no trouble executing a normal approach or flare for a touchdown at little or no vertical velocity. An investigation is in progress of the possibility of simulating the landing of the lenticular vehicle with a current fighter.

Two other flight tests mentioned will be of parachute and paraglider landing systems. Parachute systems for B-58 bailout capsule will be investigated at Mach No. 2.2. The other flight test will be to launch a $\frac{1}{2}$-size first-stage Saturn booster from a B-52 to investigate paraglider system for booster recovery.

The following recommendations for Research and Development were offered:

(1) That investigation be made to determine the effects of Reynolds number on control surfaces. Preliminary tests indicate these effects could be large.

(2) That investigations be made to determine the aerodynamic heating of control surfaces.

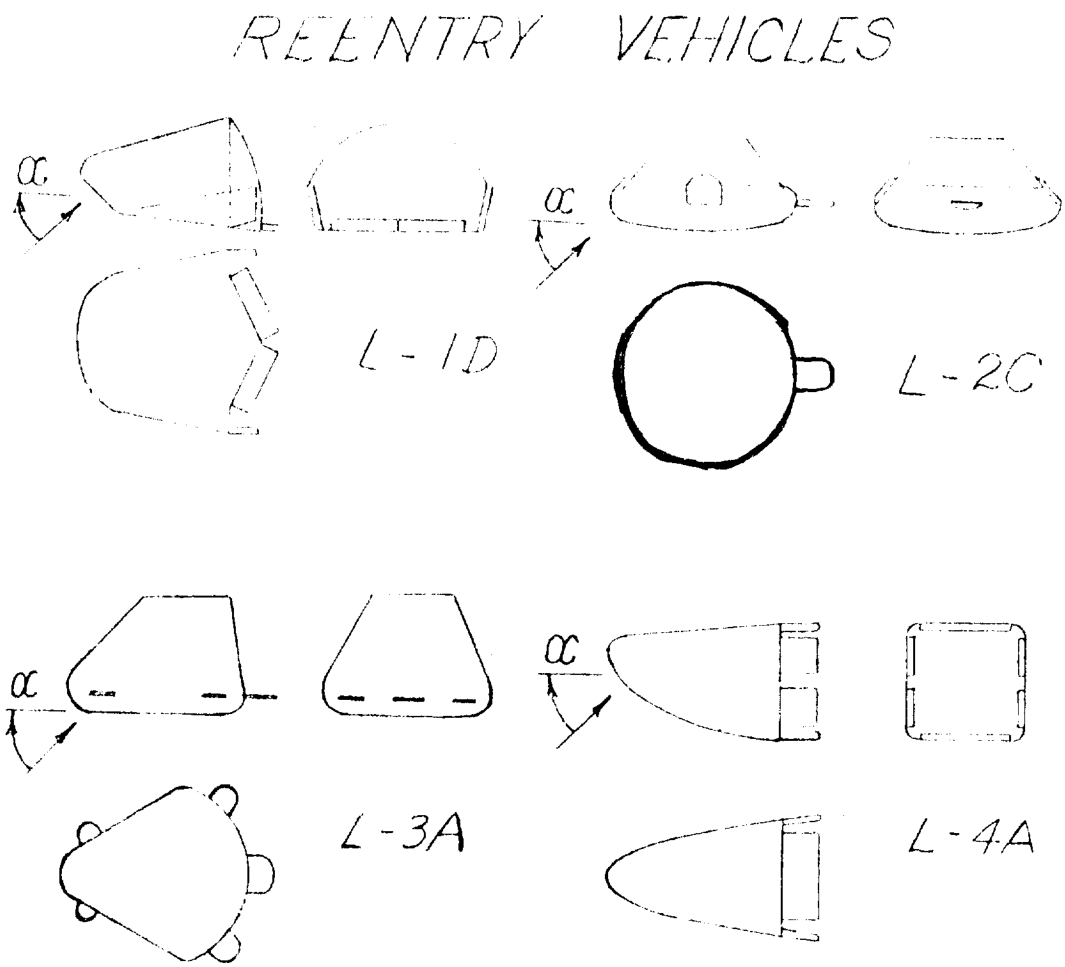
(3) That studies be made of the roll control maneuvers with center-of-gravity offset for range control.

(4) That continued effort be made to improve the water landing behavior of the lenticular configuration.

(5) That tests be made of packaging and deployment of paraglider landing systems for reentry vehicles.

(6) That tests be made of multiple parachute landing systems.

(7) That tests be made to determine the effects of jet impingement upon the static and dynamic stability.



Section VI, Figure 1.- Reentry vehicles.

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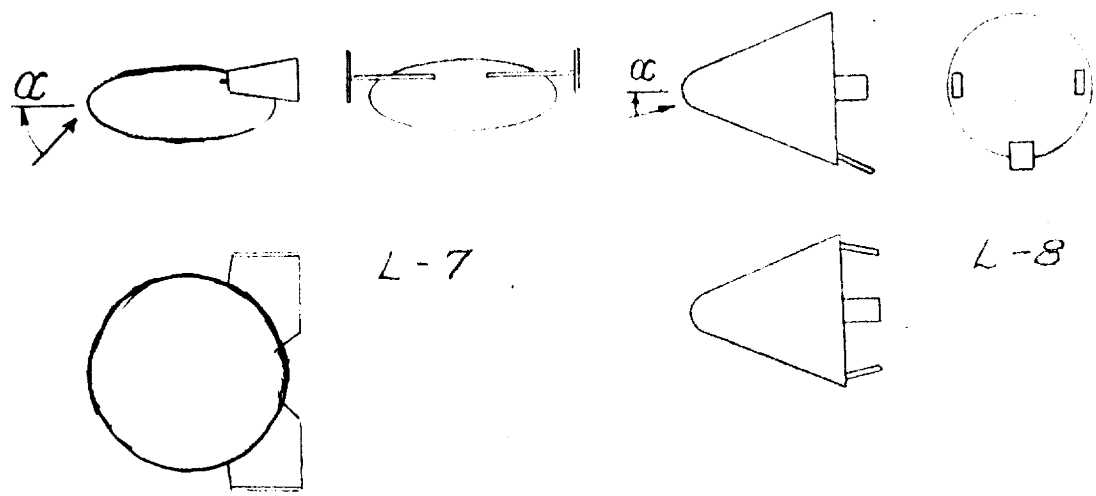
REENTRY VEHICLES



Section VI, Figure 2.- Reentry vehicles.

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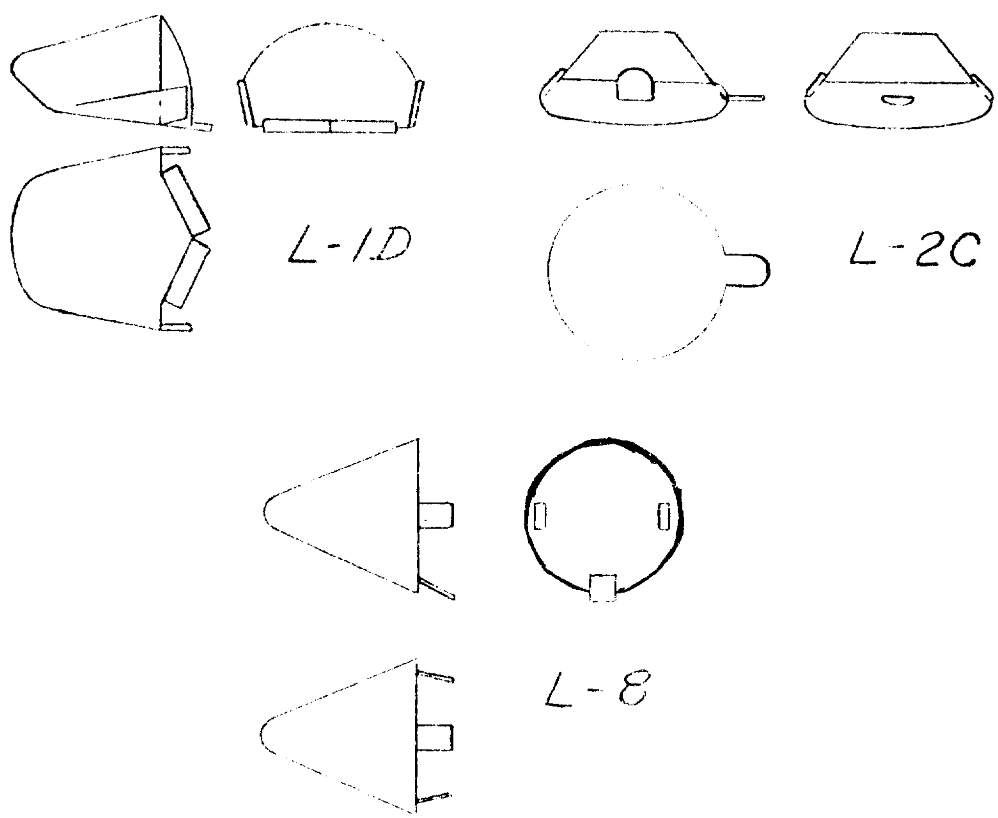
REENTRY VEHICLES



Section VI, Figure 3.- Reentry vehicles.

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SECTION VII

EXCERPTS FROM

PROJECT APOLLO MINUTES OF MEETING

OF

TECHNICAL LIAISON GROUP - HUMAN FACTORS

January 11, 1961

Ames Research Center

Moffett Field, California

[redacted]

a. <u>Ames Research Center (ARC)</u>.- Life Sciences Research to support Apollo is as follows:

The studies which have been conducted at the ARC and which have significant results of interest to the Apollo project in the life sciences field have been centered around two separate but closely related projects. One is the meaningful tolerance to acceleration studies with the pilot seated in the forward-facing position and the other is one which has to do with medical instrumentation adaptable for use in monitoring the pilot's physiological status while airborne or in a flight simulator.

There are complete reports of the work done in these projects available as NASA Technical Notes D-91, D-345, and D-351. Technical Note D-91 has to do with restraint systems for use with a pilot seated in the forward-facing position and experiencing accelerations along four different vectors, namely $-A_X$ or eyeballs out, $+A_X$ or eyeballs in, A_N or eyeballs down, and a combination of $-A_X$ and A_N acceleration or eyeballs down and out. Considerable progress has been made in the design of restraints since this initial effort which will be illustrated in a subsequent film and slides. Technical Note D-345 discusses the physiological effects of accelerations in the directions just enumerated up to a magnitude of approximately 6g and maintained for approximately 6 minutes while the pilot performed a relatively complex tracking task. Technical Note D-351 is a report on the flight evaluation of an airborne physiological instrumentation package which included some preliminary results under conditions of varying accelerations ranging from zero gravity sustained for 20 to 30 seconds to 3g during the post-zero gravity period.

Since these reports were written, there have been two other studies made which are related or are a continuation of the work described in these reports. One study is a continuation of the use of the airborne physiological instrumentation package in some flights of an F-104B aircraft. These flights were being made primarily for studies of control capabilities during zero gravity of about 60-second duration which was preceded and followed by short periods of 3g A_N or eyeballs-down accelerations. The other is a study on the Johnsville Centrifuge this past November which enabled us to further perfect instrumentation techniques as well as obtain additional objective data which was not possible during the studies on the Johnsville Centrifuge reported in TN D-345. The analysis of the data from these last two studies is not as yet complete. The study in the F-104B served to further prove the reliability of the airborne instrumentation package reported in TN D-351 and yielded further data concerning the physiological effects of varying accelerations from 0 to 3g. The November study at Johnsville permitted increasing the effectiveness of the bioinstrumentation under conditions of acceleration up to as high as 8g for 2 minutes as well as yield some rather significant visual and respiratory function data.

As the result of the studies described in TN D-345 entitled, "Physiological Effects of Acceleration Observed During a Centrifuge Study of Pilot Performance," we became interested in pursuing further these meaningful tolerance to acceleration studies. We also were interested in looking further into the areas of interest physiologically that proved to be limiting factors in this early study. These physiological areas of interest centered about the visual, respiratory and cardiovascular systems.

Because there had been reported previously that there was a visual acuity decrement during accelerations which were applied transversely or at right angles to the spinal axis and for the most part during eyeballs-out accelerations, it was decided to pursue this complaint further. The symptoms reported were those of a loss of sharpness of the image and were not those associated with changes in visual fields such as are seen during accelerations in the eyeballs-down direction. These same findings had been reported by White and others, but no reason for their occurrence had been substantiated. It was felt by us that some mechanical distortion perhaps of the cornea had taken place during the acceleration which might account for the intermittent and inconstant complaint of blurred vision. Another factor that was recognized as a possibility was that of tearing.

In order to determine if there was any corneal distortion, a technique for photographing this distortion using the reflection of a placedo disk on the cornea was used. In order to prove the reliability of this technique, a subject with 3.5 diopter of astigmatism in the right eye was photographed. Definite distortions in the reflected patterns were found by carefully measuring the distance of the rings from a central point at varying radii about 360°. Motion pictures were taken of the placedo disk reflections on the cornea during the November 1960 experiments at Johnsville, and the photographs are being examined for distortions at the present time. As yet, no distortion of the placedo disk reflection has been discovered during transverse accelerations up to as high as 8g.

As a separate approach to the same problem, a reduced scale Snellen chart was placed at a distance of 36 inches from the pilot's eye. The smallest line 20/15 was 1mm in height and this image subtended an angle of approximately 3 minutes. The light intensity of the posteriorly-illuminated chart was approximately 30-foot candles. The ability to read the 20/15 line was impaired occasionally, but most of the time all subjects could read the 20/15 line during accelerations up to 8g. Tearing again seemed to be present when blurring did occur.

Our astigmatic dial was also used which was mounted just below the Snellen chart. On only one occasion was a report made of a change in the ability to see clearly all radii of the dial.

On several occasions double vision was reported. This was true particularly in one of the subjects. A phoropter was placed in front of the right eye of this subject. This phoropter had a darkened lens in its right-eye orifice which could be rotated out and a plano-lens substituted in its stead by the pilot pulling on a string and operating a ratchet device. The pilot saw the image with the left eye only as acceleration came on. Then at a peak acceleration of 8g eyeballs out, the blocked out lens was removed and the plano substituted. A double image was seen which could never be brought into focus as a single image.

As a result of these observations, it would appear that the subjective observations of a visual acuity decrement during transverse accelerations was not borne out by the actual measurement of visual acuity while under acceleration. That no distortion of the cornea occurs resulting in an induced astigmatism is also apparent upon measurement of the placedo disk corneal reflections.

It is true that the accelerations were only prolonged for 2 minutes on the longest runs. The blurring of vision seen previously came after more prolonged exposures to accelerations of the magnitude of 6g. It is therefore possible that a vascular change in the retinal area may result which accounts for the visual decrement seen after prolonged transverse accelerations. This could be ascertained by retinal photography while under acceleration. This project is now under consideration.

The respiratory aspect of the studies conducted at Johnsville in November 1960 were devoted to a quantitative characterization of the dyspnea reported by the subjects while undergoing accelerations applied at right angles to the spinal axis of the body chest-to-back and back-to-chest.

A WEDGE bellows-type spirometer was mounted in back of the pilot's seat in the centrifuge gondola which permitted the measurement of tidal volumes, inspiratory, expiratory and vital capacities while under acceleration. It had been our intention to make nitrogen washout determinations in order to measure ventilatory indices and functional residual capacities. However, due to the failure of the nitrogen meter to function properly, these latter values were not obtained. Nevertheless, some very valuable data were obtained concerning tidal volumes and vital capacities which the subject was being subjected to transverse accelerations both eyeballs in and eyeballs out at levels of 4, 6, and 8g.

Prior to each run, ventilation volumes were measured at one g in the appropriate position by rotating the gondola placing the pilot in either the prone or supine position. It was possible under the dynamic conditions to determine these same ventilatory volumes. The measurement of the functional residual capacities was impossible because of a small leakage in the closed-circuit system. However, some qualitative observations as to residual volumes were possible. It was noted that immediately after termination of the g stress with the pilot being subjected to eyeballs-in acceleration, the pilot took several deep breaths. Assuming the spirometer regained its one g status immediately after cessation of the higher g, the shift in baseline suggested a net movement of about 400ml's out of the spirometer back into the pilot's lungs during this period of 10 to 15 seconds. A reduction of the functional residual capacities is entirely reasonable and is consistent with several other firmer observations made during the experiments. In particular, the pilots had virtually no expiratory reserve during eyeballs in g stress confirming the subjective impression that the thorax is compressed. This thoracic compression was further evidenced by the difficulty of inspiring even the small maximal inspiratory volumes achieved during eyeballs-in acceleration. The net effect of diminished inspiratory volume and zero expiratory reserve was a marked reduction in vital capacity. This, in effect, is negative pressure breathing which has been pointed out by the Aero Space Medical Laboratory at WADD and which can be counteracted by positive pressure breathing.

In the eyeballs-out direction, however, the subjects showed larger vital capacities which included measurable expiratory reserves. The mean vital capacities corrected for body surface area are shown in figure 1 for three of the test pilots. The fourth test pilot showed anomalously easy ventilation in the eyeballs-in direction which is inconsistent with the observation of the other three test pilots. This fourth test pilot's ventilatory data are included in figure 2. Pilot number four is of an entirely different somatotype from the other subjects, being 6 feet 2 inches tall and weighing but 152 pounds.

It is interesting to note that the minute volumes actually increased at 6g and 8g eyeballs out when compared to the 1g level as shown in figure 3. This may be of considerable significance. Despite the reduced minute volume eyeballs-in acceleration, the periods of time were too short for the development of hypercapnia which could account for the massive drive to increased ventilation that the subjects showed - a drive that carried over into the poststress period as is evidenced by the hyperventilation at that time. It may be that transverse g stress causes a tonic stimulation of receptors in the lung and perhaps the chest wall which are sensitive to deflation. Although the dyspnea is less marked in the eyeballs-out direction, the increased minute volume at high g suggests an inspiratory drive which, under these easier conditions of ventilation, can actually be indulged by the pilot. In this slide you will note there is plotted the ventilation of a naive subject whose hyperventilation was clearly due to apprehension; nevertheless, the pattern of increased ventilation in the eyeballs-out direction is still apparent.

Figure 4 shows the mean tidal volumes achieved at each level of g stress. The tidal volume is not very informative datum <u>per se</u>, but the smaller volumes with increasing g stress in the eyeballs-in direction are consistent with the more clear-cut changes in vital capacity.

The cardiovascular findings of note in recent experimentation involve the blood pressure findings obtained in flight during zero g parabolic maneuvers in the T-33 as well as the F-104B. A consistent fall in diastolic pressure is noted during the zero g period with little change in systolic pressure. During the F-104B flights, ventricular extra systoles are noted during the postzero g period of 3g acceleration, (A_N), which were not noted during the prezero g period of 3g acceleration on two flights. It might be significant that these extra systoles occurred during the postzero g 3g acceleration and illustrates some intolerance to acceleration after a period of zero g. Time histories of the heart rate, respiratory rate and blood pressures are shown in TN D-351 for the T-33 flights and in the "handouts" for the F-104B flights.

Work which is planned for the near future includes:

(1) Angular acceleration threshold studies to be made by Mr. Brant Clark on the ARC three-degree-of-freedom simulator and possibly in aircraft.

(2) Johnsville Centrifuge studies in March and April will cover the following items:

(a) Performance-type tolerance versus time-acceleration studies.

(b) Post-acceleration performance studies using an ILS approach task.

(c) Side arm controller evaluations plus ventilation studies to include nitrogen washout studies giving ventilatory indices, both for time and nitrogen levels at time of leveling off; functional residual capacities; CO_2 levels; tidal and minute volumes in eyeballs down, eyeballs out, and eyeballs-in directions of acceleration.

(d) Effect of rate of onset on pilot performance.

(e) Performance during drag-modulated entries including effects of rate of onset.

(3) Medical Instrumentation:

(a) In-flight telemetry of physiological data. (Subject to vehicle to ground.)

(b) EEG transducers and amplifiers. Dr. Noel Thompson, PAMRF.

(c) Pending proposals from Southwest Research Institute and Corbin and Farnsworth for indirect recording of blood pressure without use of the cuff method and a vectocardiographic read-out system giving instantaneous evaluation of cardiac malfunction on the basis of a real-time display of magnitude versus time-and-phase angle versus time of the electric cardiac vector.

(4) Restraint System for omnidirectional acceleration stress which is primarily directed at sustained accelerations, but will be investigated from the standpoint of impact support.

(5) Vision Problems in Transverse Acceleration - Would like to fully evaluate the problem. Dr. Tom Duane is interested and would like to do some retinal photography.

(6) X-ray Studies of Thoracic Contents - Cinematography using millisecond exposures at 2,000 amperes is being contemplated by Dr. Earl Miller, Head of the Department of Roentgenology, University of California.

Dr. Smedal also described the extensive simulation facilities available at ARC including the present two- and three-degree-of-freedom cockpits as well as the large air-bearing simulator and the five-degree-of-freedom centrifuge currently under construction.

ARC has been doing work on a "portable" omnidirectional pilot restraint system. The pilot is laced into his "seat" which is articulated. He can connect it to fastenings in the simulator or spacecraft for restraint or uncouple to move about. Present designs are somewhat bulky, but work is continuing to improve this situation.

b. Flight Research Center (FRC).- Apollo-related efforts are as follows:

Investigations pertaining to Human Factors have been grouped in three main categories. These are: (1) Determination of pilot control

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capability boundaries and the factors influencing these; (2) Physiological; and (3) Display. Most of the current work involves the X-15 and the analog simulator. Future work is planned which will involve other research aircraft such as the F-104 and the F-100C variable stability airplane.

The determination of pilot control capabilities and the factors influencing these is accomplished with three research tools: X-15, F-100C variable stability airplane, and the analog simulator.

(1) X-15 - Quantitative and qualitative evaluation of pilot performance during flight while subjected to the varying acceleration profile will be undertaken. This can consist of longitudinal accelerations from 2g to 4g approximately, prolonged zero g, and high-sustained normal g. The moderate level of longitudinal acceleration is similar to levels expected in some boosted trajectories. Performance in flight will be compared to performance in the fixed-base simulator for the same missions. The new, larger engine will permit flights with zero g duration of from 4 to 5 minutes as well as higher positive g levels.

(2) F-100C - Variable Stability Airplane - Determination of the minimum acceptable control requirements for X-15, DS, and other future vehicles from the pilot adaptability aspect. This will involve development and utilization of various control techniques.

(3) Analog Simulator - Determination of minimum control requirements for a wide range of configurations and development of control techniques. In addition, examination of control systems to improve control capability through variation in system response, controller design, etc., the various phases of orbital flight (boost-cruise-reentry) will be examined to define pilot capability and performance.

Physiological investigations include the X-15 and other research airplanes. A physiological package is presently in use in the X-15 to record: (1) helmet to suit differential pressure, (2) suit pressure, (3) cabin temperature, (4) pilot's body temperature, (5) respiration rate, (6) pulse rate, (7) EKG, and (8) oxygen flow rate. Total oxygen consumption is also obtained from preflight and postflight quantity measurements. Quantitative measurements of pilot performance under flight conditions will be obtained by comparison with simulator performance. Cosmic radiation instrumentation will be installed externally on the X-15 cockpit. Other investigations are planned to evaluate another physiological package in an F-104 which will measure all the previously-mentioned quantities in addition to blood pressure. Both physiological packages will be used in the simulator prior to flight to establish a baseline performance level. An investigation of the effects of spatial environment, (zero g), on pilot perception of simultaneous

[redacted]

orthogonal angular rates is planned in an F-104 for comparison with results obtained in ground-based simulators.

The display investigations will include quantitative evaluations of scanning procedure variations and instrument utilization during X-15 flights. An energy-management display development is planned for use in the X-15. Display quickening will be evaluated in an F-104 reaction control airplane and the F-100C variable stability airplane to evaluate the performance improvements. Provisions for displaying attitude rate, $\dot{\alpha}$ and $\dot{\beta}$, etc., will be incorporated. Tape-type α and g indicators will be used in the reaction control F-104 airplane. An analog investigation of display requirements is planned which will include evaluations of present systems. The effects of quickening on controllability boundaries will be investigated on the simulator.

FRC is most interested in the contact analog visual display system developed under the Army-Navy Instrumentation Program of which one version is being installed in the A2F-1 airplane, another in the Shark nuclear-powered submarine. FRC pilots are planning to test-fly a version of this display which is being installed in an R4Y by NADC, Johnsville.

FRC will do some testing of adaptive control systems, probably in the F-100C variable stability airplane and in simulators. The systems will adapt in the sense that gains will be adjusted automatically as a function of the dynamic response of the aircraft-to-control inputs.

FRC is interested in pressure suit development work.

c. Lewis Research Center (LeRC).- Work applicable to Apollo Human Factors is as follows:

At the present time, work in the Human Factors area at LeRC is limited to literature reviews for the purpose of accumulating engineering design information. A paper is being prepared relative to radiation levels in space and the shielding thicknesses required. Estimates of shield weights for various type missions are included. Another paper in preparation summarizes information about meteorite size and mass distribution, frequency of impact, depth of penetration, etc. Another effort involves collecting information as to food and water requirements, waste disposal, water recovery, respiratory requirements, favorable temperature and humidity conditions, etc.

d. Langley Research Center (LRC).- Studies relating to Apollo are as follows:

The following theory was developed by Messrs. Adamson and Davidson:

One of the greatest dangers confronting the astronaut is posed by the chance of his encountering a solar flare event. These events, though conforming to the same general pattern, vary markedly in scale. The types of solar event that pose the major hazard are the so-called high-energy events of the high-flux, low-energy events.

High-energy events occur, on the average, once every 5 years. They are composed of protons ranging in energy up to tens of Bev. The flux of particles being of the order of hundreds of thousands per cm^2 per second.

High-flux events occur more frequently; thus, during the past solar maximum, as many as four have been observed in a single year. The protons involved in such an event have energies in the range of 20 Mev - 500 Mev. However, the fluxes may amount to millions of particles per cm^2 per second.

If we assume these events are randomly distributed, we can assess the probability of our encounters. For example, in an event during a 10-day mission for a high-energy event, the chances amount to one in 200 which appears to constitute an acceptable risk. With regard to the high-flux events, by virtue of their much greater frequency, the chances amount to about one in 10. This is almost as risky as playing Russian Roulette, and it is the feeling that even astronauts would balk at exposing themselves voluntarily to risks of this magnitude.

These probabilities are based on the supposition that the events are randomly distributed in time. A very superficial examination of the data serves to convince us this is not the case. For one thing, the events exhibit a tendency to bunch together for which due allowance should be made.

Within the past few months at LRC an attempt has been made to determine the statistical characteristics of proton events with a view to obtaining more realistic estimates of the probabilities involved. Before describing the details of the analysis, a little general background should be provided. The sun is known to have a dipole magnetic field (fig. 5). Moreover, since the sun is a hot plasma, the lines of force are firmly anchored to the surface (slippage is forbidden). The dipole field must participate in the sun's rotation. The situation is further complicated by the fact that the sun is continuously emitting plasma, as illustrated in figure 6. Consider the emission of a single lump of plasma. Bear in mind that to all intents and purposes, relative motion between plasma and magnetic lines is forbidden; hence, plasma must be dragged around by the rotating dipole field and a centrifugal force is operative on it. In addition, the plasma drags magnetic lines

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out of the sun itself. These lines behave like elastic strings and will apply a small radially inward force. There is, therefore, a resultant force tending to draw the plasma into the sun's equatorial plane. In the light of these considerations, we deduce that the dipole field becomes distorted as shown in figure 7.

From time-to-time, intense flares appear on the sun's surface. Their mechanism of origin is far from thoroughly understood. However, it is known that they are accompanied by ejection of a proton stream which reaches the earth's orbital radius in about half an hour, and emission of a plasma cloud which reaches the earth's orbital radius in about 24 hours. Both the proton stream and plasma cloud will tend to follow interplanetary magnetic lines. As a result, there is good reason to believe that when one engulfs the earth, the chances are the other will also. This is borne out by the data presented on figure 8.

The geomagnetic index is a measure of the disturbance of the earth's magnetic field which is known to result from the impingement of the plasma cloud. The proton stream in principle distorts the geomagnetic field also. However, by virtue of its even smaller flux, its contribution is entirely negligible. The riometer index on the other hand is a measure of the disturbance in the ionospheric shells resulting from the impingement of the proton stream.

Now, we come to the critical point in the entire analysis. It is the proton stream which passes the radiation hazard and we are, therefore, specifically interested in their statistical characteristics. However, riometer data extends only over the past 3 years and does not provide an adequate basis for statistical evaluation. The good correlation existing between geomagnetic storms and proton streams, to which allusion has already been made, encourages us in the belief that the statistical characteristics of geomagnetic storms will resemble closely those of proton streams. It is on this basis we proceed.

As shown in figure 9, the concentric rings are associated with the different years of the preceding solar year (from 1943 to 1953). Around the periphery are the different months of the year. The major geomagnetic storms occurring in the preceding cycle are represented by radial line segments. It is clear at a glance the geomagnetic storms show a propensity to occur during the spring and autumn months (i.e., March, April; September, October). This seasonal variation has long been recognized, however, by virtue of the correlation previously established. It follows that proton streams are encountered in the earth's proximity most frequently during these months. Such a seasonal variation is to be expected in the light of the theoretical picture we have already drawn of the interplanetary magnetic field having a disk-like structure. Thus, as the earth moves in an orbital plane which is inclined to this disk, such a seasonal variation would follow.

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If our reasoning is valid, then by launching during favorable months (midwinter or midsummer), the chances of our encountering a high-flux event (assuming, as before, an average of 4 per year) during a 10-day mission are reduced from one in 10 to one in 20.

Reverting to figure 9, in addition to noting a seasonal variation, we note a tendency for the events to bunch together. This will tend to widen the gap between successive sequences and will further lessen chances of our encountering an event on a 10-day mission. We find, in fact, when due statistical allowance is made for this, the chances of our encountering a single event are further reduced to 1 in 24.

Though bunching of events reduces our chances of encountering a single event, it increases the chances of our encountering two or more events. Again, taking four events per year and a 10-day mission, the probability of our encountering two or more events assuming a Gaussion distribution is about 1 in 160. When allowance has been made for bunching, this probability becomes 1 in 80.

All of the probabilities quoted previously are based on the occurrence of four events per year which appears to be representative of the previous solar maximum. Solar activity is, however, subject to both an 11-year periodicity and a longer periodicity (about 90 years).

Indeed, if we plot solar activity as a function of year, we obtain the results shown in figure 10.

Drawing a line through successive solar maximum, we obtain a saw-toothed curve. The past solar maximum seems to be just at the top of one of the sawteeth and the following maximum 1967 to 1970 may possibly be substantially less. There are indications that the past cycle has been particularly erratic which might well suggest an incipient instability preceding a major change in solar activity level. It is my own feeling that this should not be relied upon.

Dr. Foelsche of LRC is making computations of radiation dosages as a function of shielding thickness (water) associated with some of the more intense solar events. These computations are based on the following three assumptions:

(1) That the solar flare event comprises only protons

(2) That the time variation of the energy spectrum is known

(3) That the production of secondaries within the material can be disregarded

The validity of assumption (3) is presently being investigated at LRC.

e. Office of Life Science Programs (OLSP).- Grants and contracts applicable to Apollo are as follows:

OLSP has divided its efforts into three principal categories:

(1) Flight medicine and biology

(2) Space medical and behavioral science

(3) Space biology

In the first category, the following contract efforts may produce results applicable to the Apollo program:

(1) Study of KO_2 and NaO_2 - Mine Safety Appliances.

(2) Recovery of O_2 from CO_2 - Isomet Corporation.

(3) Purification of H_2O from wastes - General Electric Co.

(4) Investigations of various trace contaminants given off by the human body - Southwest Research Institute.

(5) Analysis of biodesign principles for application to instrument design - Bell Aircraft Corp.

(6) Management contract for a research program leading to development of an Integrated Life Support System - not yet contracted.

(7) Management contract for a research program entitled "An Integrated Performance Study," to include display, control, sensing and computation requirements to integrate man and machine for both manned and remotely controlled space flight - not yet contracted.

Although the last two studies are not specifically mission-oriented, they may possibly produce some results applicable to the Apollo system.

All of the work in Space Medical and Behavioral Sciences is believed applicable in some degree to the Apollo program. Contracts most directly applicable are:

(1) NASA participation in Bio-sciences Information exchange. Contracted to Bio-sciences Information exchange.

(2) Effects of angular velocity, angular acceleration, coriolis acceleration and tumblings - U.S. Navy (Pensacola, Fla.).

(3) Bibliography of Life Sciences for Space - Library of Congress.

(4) Biological experiments on the effects of radiation in the upper atmosphere using high-altitude balloons - U.S. Air Force.

(5) Feedback information criteria for functional extension of human hands - Massachusetts Institute of Technology (MIT).

The contracts in the field of Space Biology most applicable to Apollo are:

(1) Biological systems in space - University of California

(2) Sterilization of vehicles - U.S. Army

In addition to the grants and contracts, OLSP is planning a number of in-flight experiments, using experimental animals ranging from unicellular to primate. There will also be subcellular experiments. Many of these experiments should provide information valuable for Apollo.

f. Space Task Group (STG).- STG work in Human Factors for Apollo is as follows:

(1) Consideration of Project Mercury medical support experience which may be applicable to Project Apollo.

(2) "Project Apollo Life Support Programs" is a series of suggested research proposals applicable to Apollo.

The Project Mercury crew support experience has provided bioscience data applicable to Apollo in several areas. Mercury crew selection and training procedures have evolved certain selection guidelines and developed simulation concepts which may have advanced-vehicle application. Physical fitness, a record of proven performance under stress, intelligence, and adaptability are obviously desirable crew characteristics found in Mercury Astronauts. An attempt is being made in Mercury to quantitate crew stress by hormone and psychological measurements. Such data may have value in crew selection techniques. Physiological measurements, assessing cardiopulmonary functions during astronaut centrifuge runs, point up the need for much additional data to safely support the relatively long duration, moderate magnitude g forces expected in Apollo.

Detailed physiological control data acquisition and meticulous preflight and postflight astronaut examination will provide information on the effects of combined flight stresses seen in the ballistic and orbital flights. Evaluation of the physiological-psychological effects of weightlessness should be particularly interesting.

Bioinstrumentation considerations in Mercury provide research and development direction toward advanced biosensors and environmental monitoring devices. Respiration sensors with volume and flow capability, noninterference blood pressure sensors, blood gas monitors and flight electroencephalography appears desirable and feasible.

Medical monitor planning considerations have indicated the need for additional reliable real-time man-monitoring equipment.

(1) It was noted that the use of experimental test pilots in the conduct of Apollo studies should be strongly considered.

(2) It was noted that Batelle Institute is gathering information on radiation data under a U.S. Air Force contract and that the quarterly reports contain useful information.

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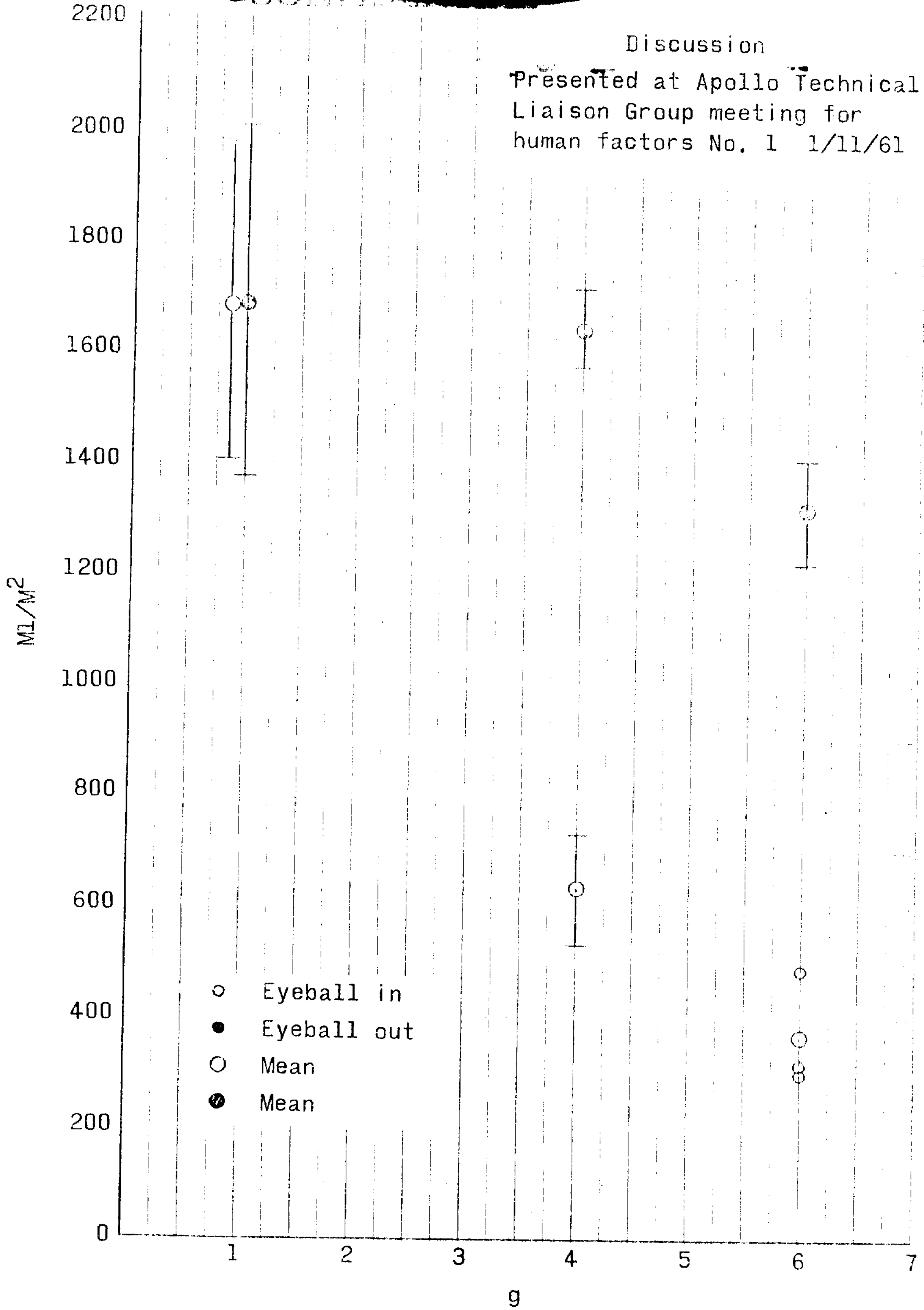

Section VII, Figure 1.- Mean vital capacities, Ml/M^2 $\pm$SD.

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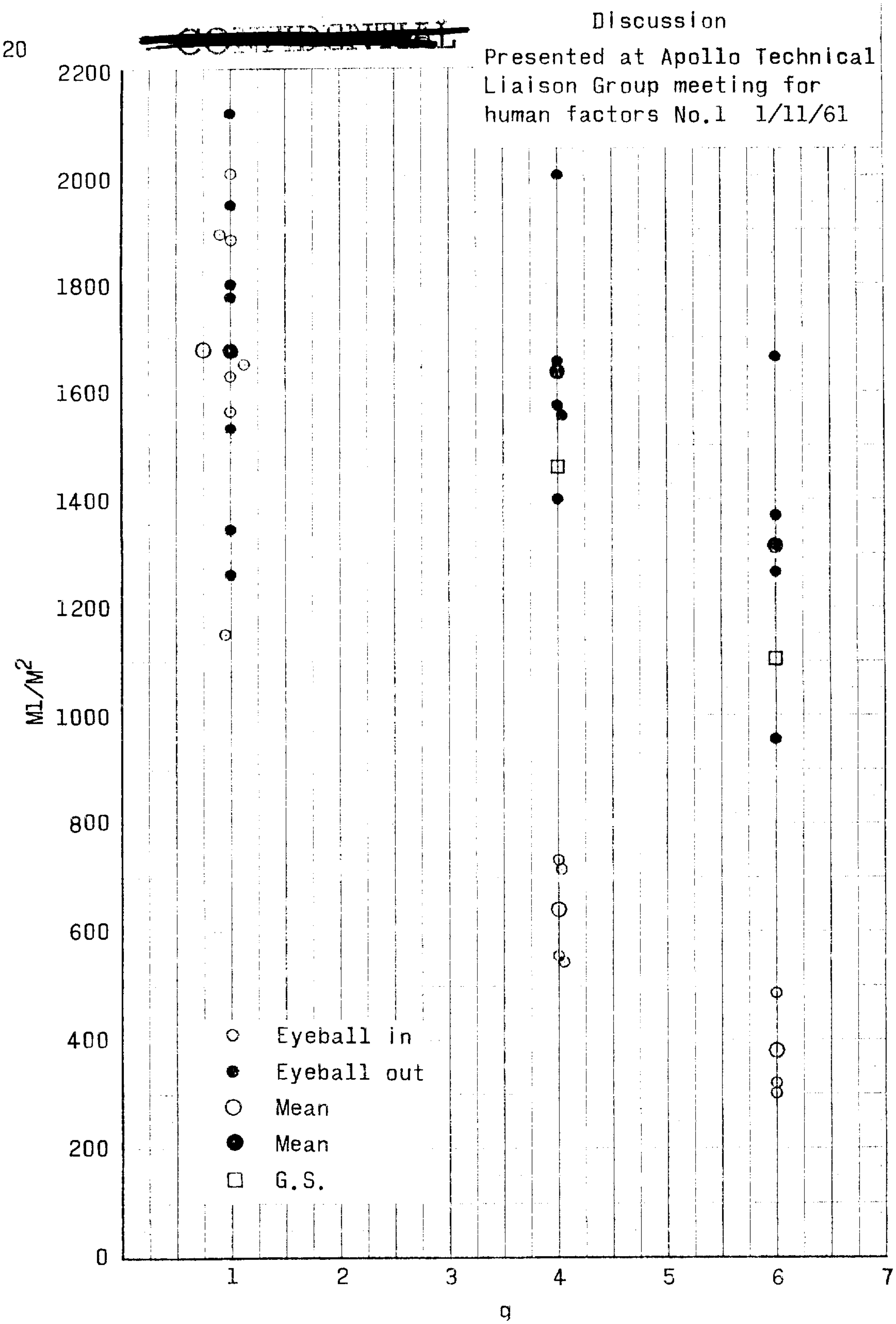

Section VII, Figure 2.- Vital capacities, Ml/M^2.

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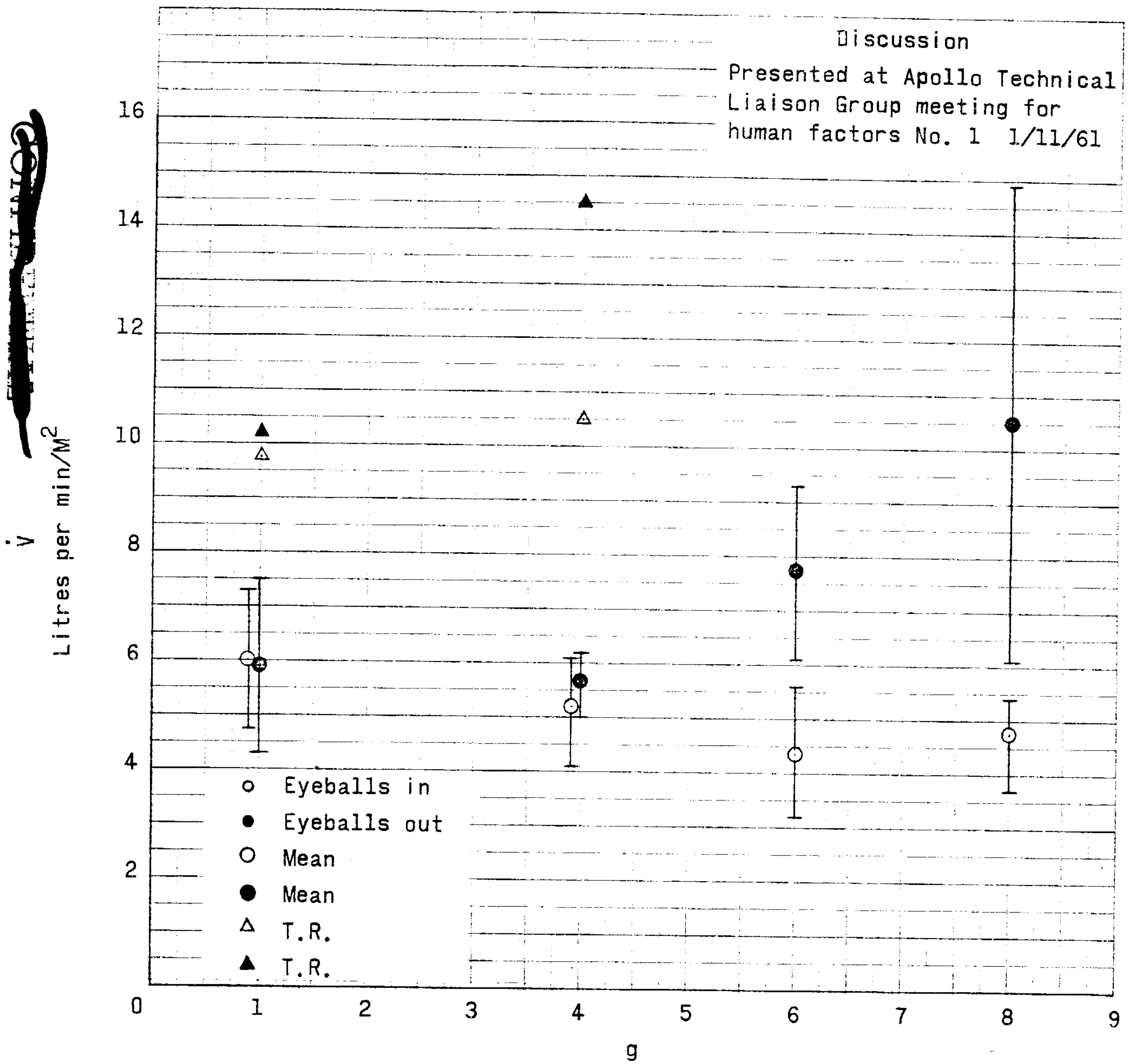

Section VII, Figure 3.- Mean minute ventilation volume ±SD. litres/M^2.

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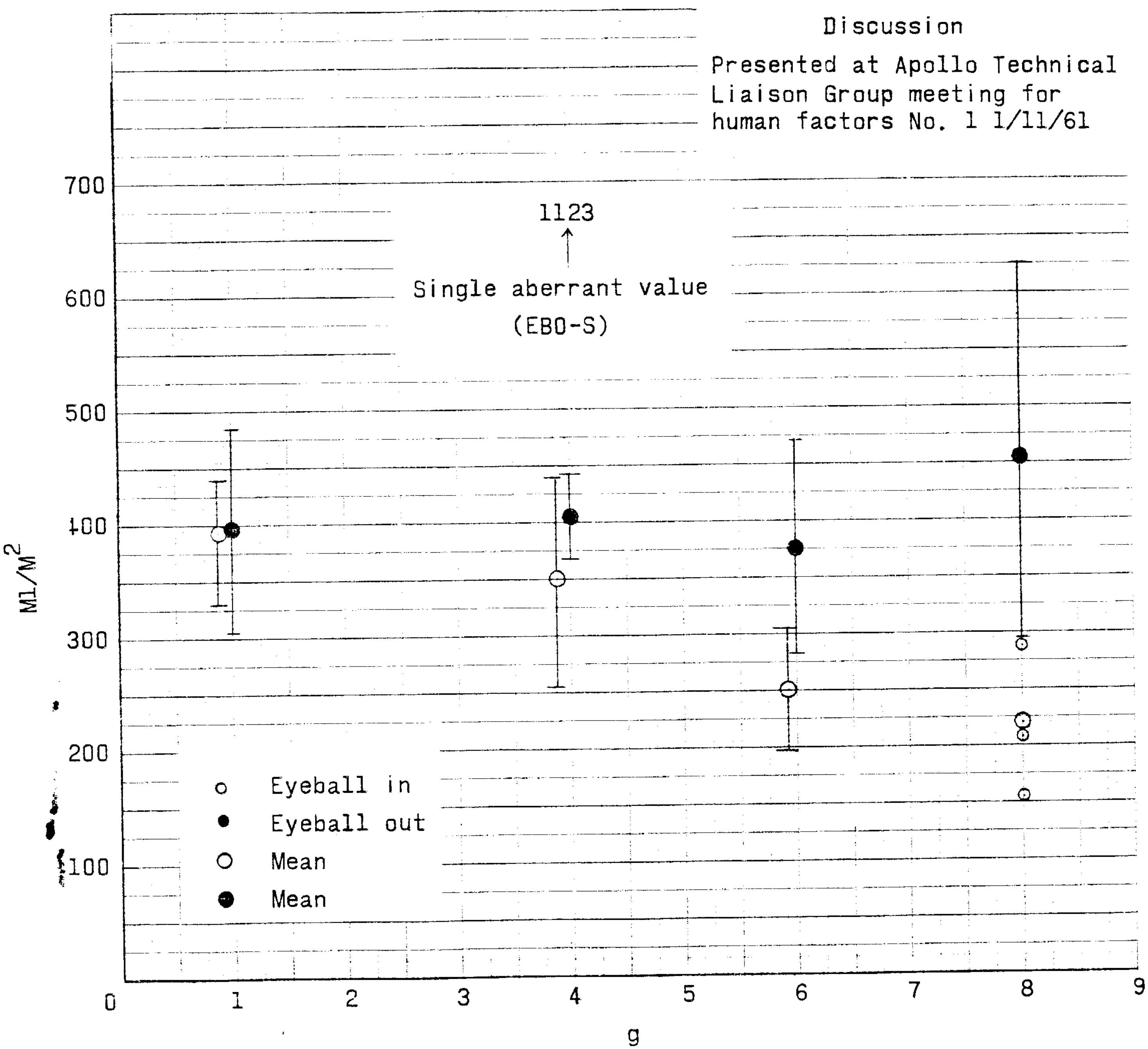

Section VII, Figure 4.- Mean tidal volumes, Ml/M^2 $\pm$SD.

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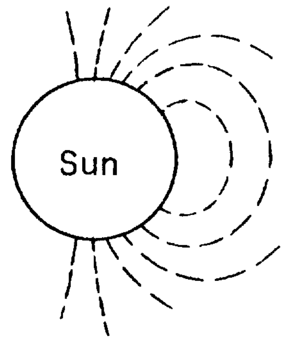


Figure 5.

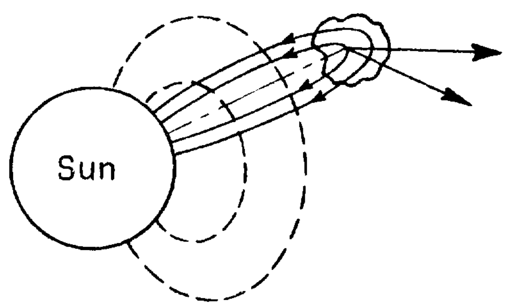


Figure 6.

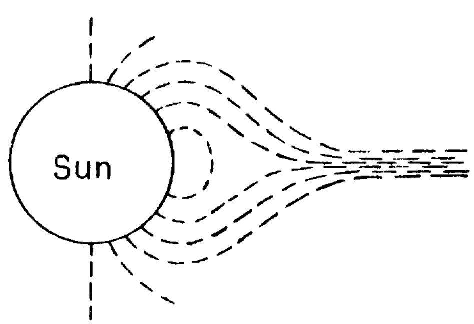
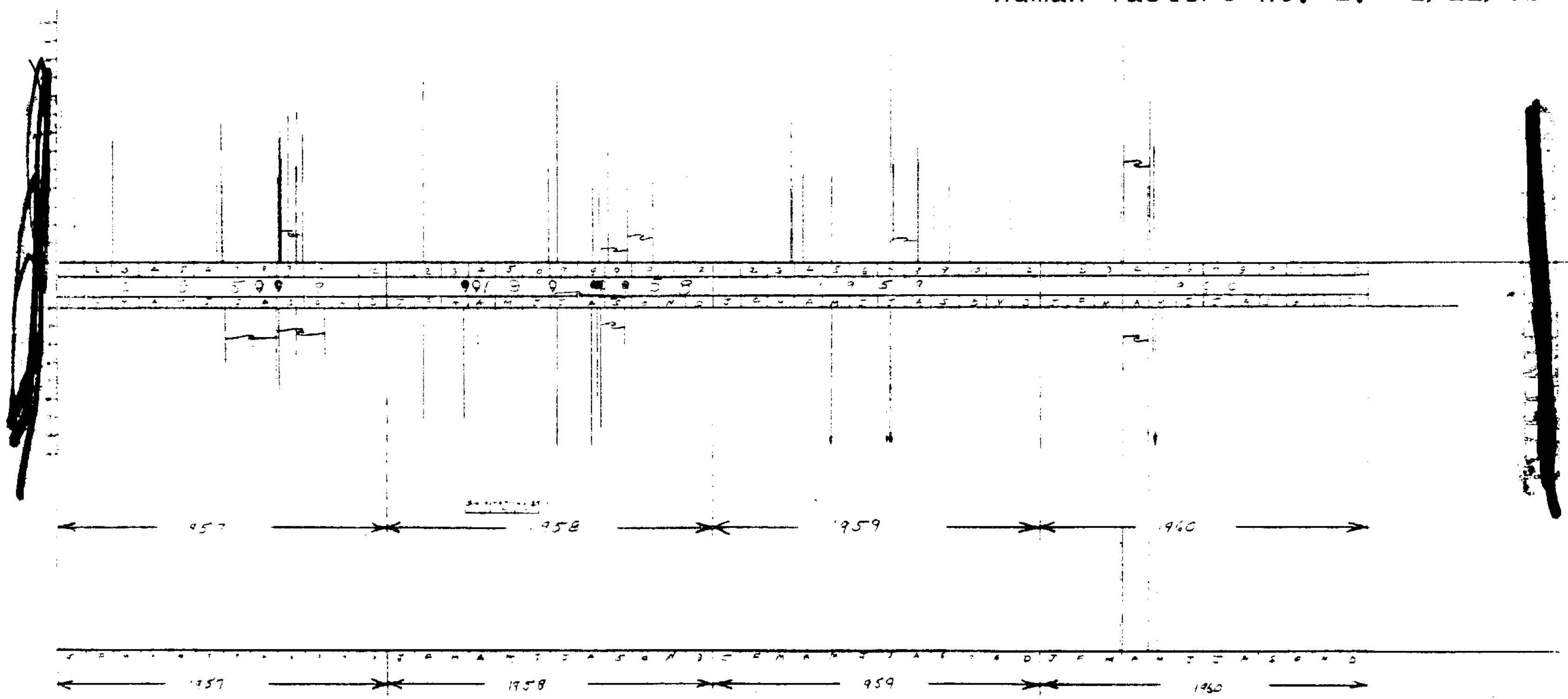


Figure 7.

Section VII, Figure 5, 6, and 7.- Discussion presented at Apollo Human Factors Technical Liaison Group Meeting No. 1 - 1/11/61.

Discussion
Presented at Apollo Technical
Liaison Group meeting for
human factors No. 1. 1/11/61



Section VII, Figure 8.- LRC discussion presented at Apollo Human Factors Technical Liaison Group Meeting No. 1 - 1/11/61.

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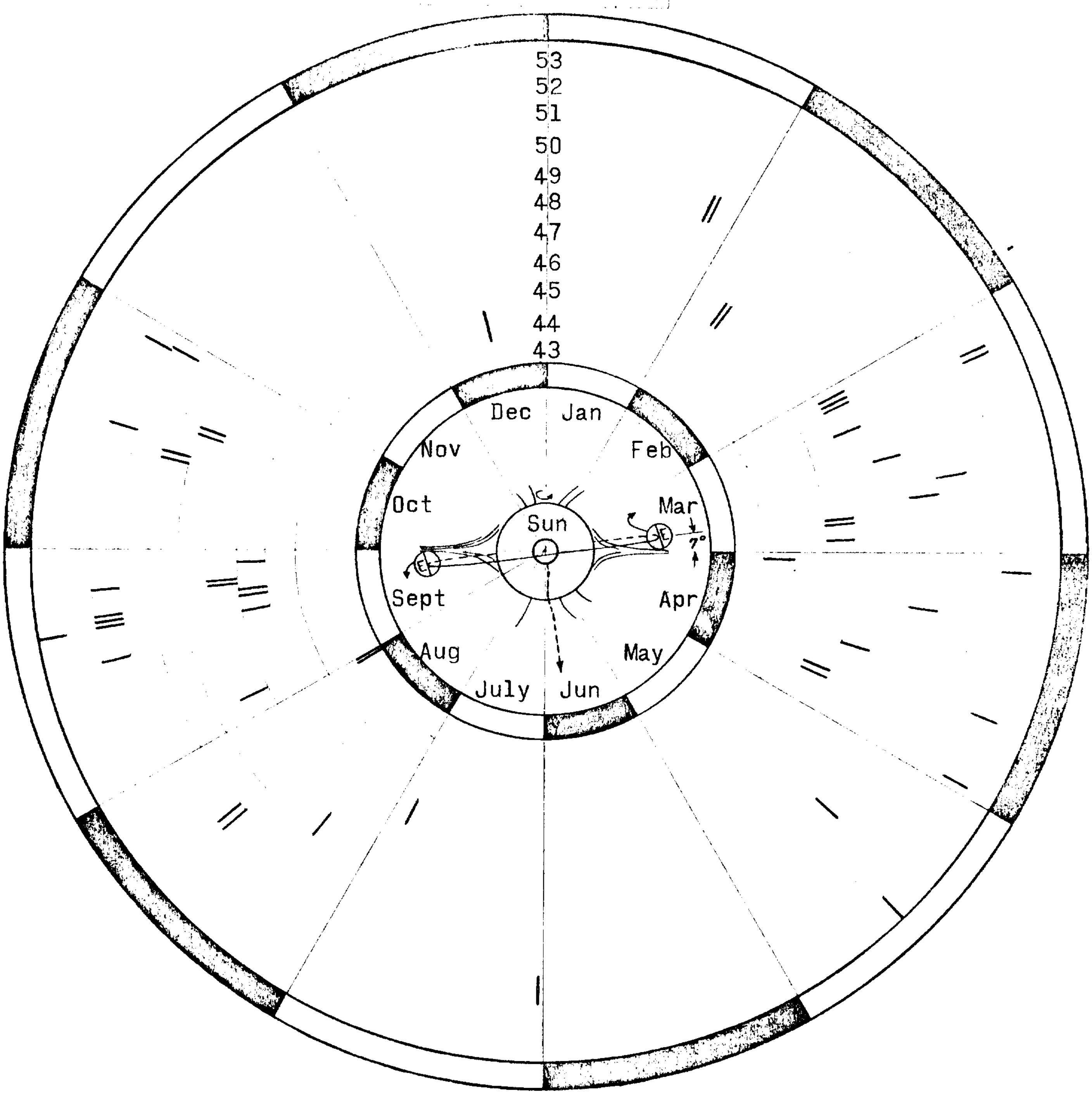

Section VII, Figure 9.- LRC discussion presented at Apollo Human Factors Technical Liaison Group Meeting No. 1 - 1/11/61.

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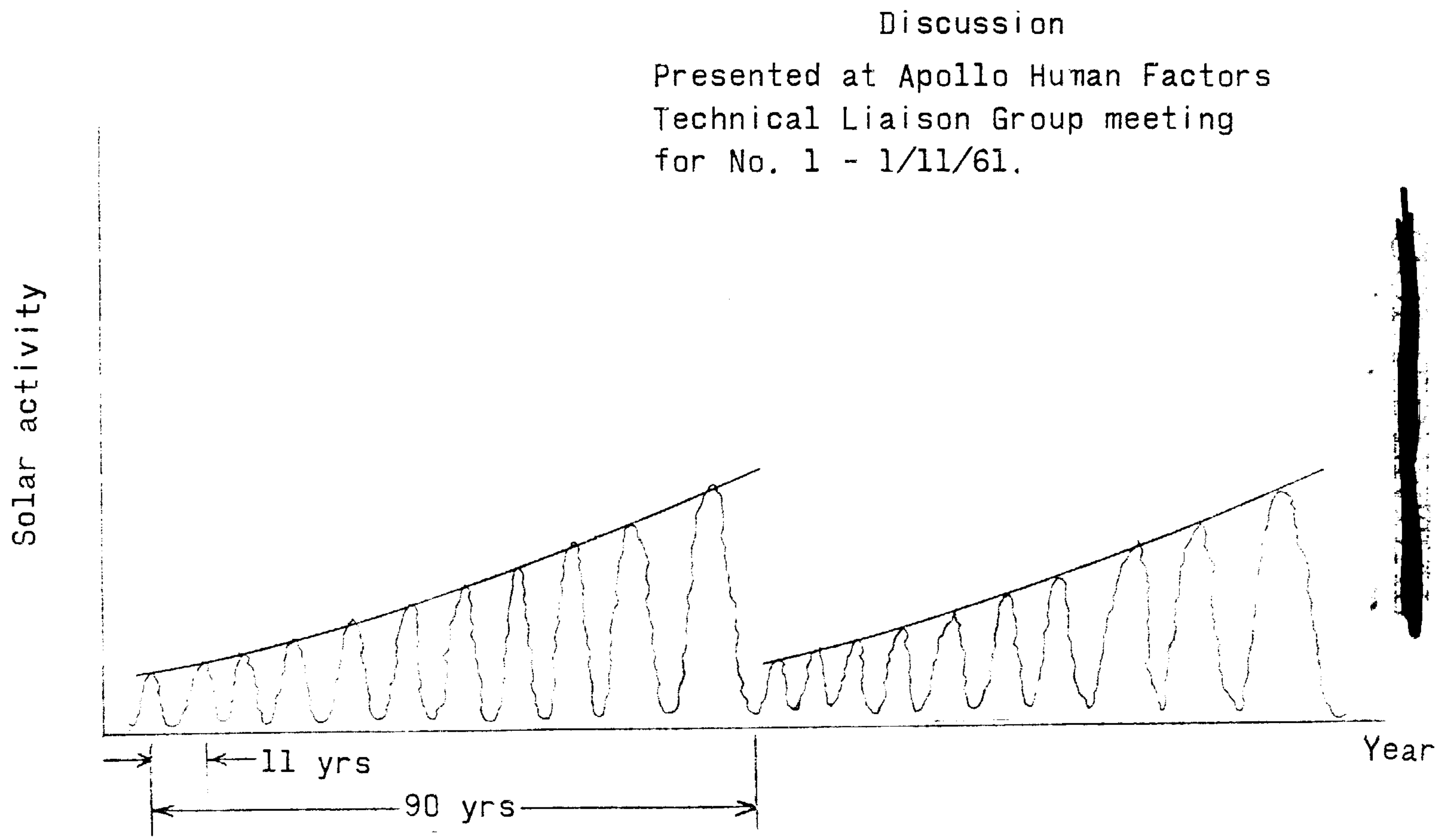

Section VII, Figure 10.- Discussion presented at Apollo Human Factors Technical Liaison Group Meeting No. 1 - 1/11/61.

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SECTION VIII

EXCERPTS FROM

PROJECT APOLLO MINUTES OF MEETING

OF

TECHNICAL LIAISON GROUP - ONBOARD PROPULSION

January 6, 1961

Space Task Group

Langley Field, Virginia

[redacted]

a. Space Task Group (STG).- The STG has recently completed a study entitled "Preliminary Survey of Retrograde Velocities Required for Insertion into Low Lunar Altitude Orbits." (NASA Project Apollo Working Paper No. 1005). A proposed working paper entitled "Preliminary Analysis of the Propulsion Requirements for Mission Abort at Earth Escape Velocity" is near completion. This analysis covered thrust to weight ratio (T/W) from 2 to 4.5. Mr. Hammack stated that similar studies at Marshall Space Flight Center (MSFC) showed the optimum T/W was between 2.5 and 4.5 for mission abort at escape velocities.

b. Langley Research Center (LRC).- LRC activities on propulsion systems applicable to the Apollo program are confined to analytical and experimental studies leading to very high mass fraction solid-propellant rocket motors, and to advanced design and fabrication concepts which make it possible to produce these rockets with a minimum effort. Objectives of these studies are minimizing case, insulation and nozzle weight, minimizing sliver, and maximizing loading density, while still retaining excellent internal ballistic characteristics. When homogeneous, isotropic case construction materials are used, minimum case weight for a given operating pressure and case volume is achieved by using a spherical shell. For a number of years, LRC has been designing, producing, and using spherical solid propellant rockets in its own flight programs. While a majority of this work has been in-house, some work has been carried out on contract. Spherical rockets have been successfully fired having the following characteristics:

Diameter	5 to 40 inches
Weight	4 to 2,100 pounds
Specific impluse	200 to 270 seconds
Propellant weight faction	0.89 to 0.94
Operating pressure	600 psi

All of these rockets utilized metal cases which were constructed of commonplace metals (4130 steel and 2024 T6 aluminum) which had very modest yield strengths (150,000 to 180,000 psi for the steel). At present, work is underway for development of a 15-inch diameter spherical rocket which would have a titanium case and which would result in a rocket having a propellant weight fraction greater than 0.95.

Studies have been made on filament-wound fiber-glass cases which would use the same charge design philosophy used in the spherical rocket. These studies show that the minimum weight pressure vessel constructed of this material will be a shape which approximates an oblate spheroid.

Suitable charge designs have been made and have been successfully fired in heavy weight still cases. Design and performance parameters of these charges were as good or better than those obtained with the spherical charges. Contracts are being let for the development and production of filament-wound fiber-glass shells, which will allow the development of solid propellant rockets which have a propellant weight fraction in excess of 0.96 in relatively small sizes.

Other research is aimed at improving methods of fabrication of the intricate charges needed for spherical and spherical rockets, and which are applicable to all solid-propellant rocket designs. One approach is the use of an extremely lightweight foamed plastic (2 lb per cu ft) for construction of the mandrel. The igniter is designed as an integral part of this mandrel. Propellant is cast between the mandrel and the motor case, secured in place, and the motor is fired without removal of the mandrel. This allows the use of charge and case designs not possible before as the mandrel does not have to be withdrawn through a finite-sized opening, and does not have to have a constant or uniformly varying cross-section with respect to the rockets' longitudinal axis. Numerous M-58 rockets have been successfully constructed and fired using this technique. Work is currently underway to adopt this manufacturing process to spherical motor designs. LRC is planning to install a pressure-fed system for handling storable propellants. This system will be available for evaluation work but is not intended for development work.

c. Marshall Space Flight Center (MSFC).- MSFC activity directly related to the Apollo propulsion system has been concentrated in determining the midcourse and abort propulsion system requirements based on the Saturn trajectories:

Studies are still in progress; however, the results obtained to date show the trends to be expected.

Figure 1 shows a typical example of how the thrust-to-weight ratio influences the amount of total spacecraft weight which must be provided as propulsion system for abort just prior to injection on the lunar mission. These curves are for a specific injection altitude.

Figure 2 shows the influence of injection altitude and injection angle on the velocity requirements for abort at injection velocity.

Studies are also in progress on the abort requirements earlier in the trajectory. The critical requirement occurs early in third-stage burning and is due to the relatively steep ascent angle of the trajectory at this point which results in a steep uncorrected abort reentry angle which would result in decelerations up to 20g to 25g. Flattening the

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trajectory and lowering the injection altitude tend to decrease this requirement.

Indications of the present studies are:

(1) Payload increases as the injection altitude is lowered to limits imposed by heating of the payload or vehicle structure. These do not appear serious at 90km and above but a lower limit has not been investigated.

(2) Payload decreases as injection angle is forced downward from 2° above the horizontal; however, the decrease is not serious for small angles (up to 2° below the horizontal) and the abort thrust-to-weight ratio and required propellants decrease rapidly.

(3) Critical abort conditions prior to injection occur early in third-stage burning but are nearly as severe over a range of burning time from late second-stage burning through about the first $\frac{1}{3}$ of third-stage burning. Lowering the injection altitude and flattening the trajectory reduces this requirement for abort propulsion.

(4) Presently it appears that T/W ratios in the neighborhood of 2 to 3 and velocity increments of about 800 meters per second will be adequate for the abort propulsion, and injection conditions near horizontal at 90km will provide the most desirable conditions. It should be noted that these studies are still in progress and the above conclusions are tentative.

A brief summary of MSFC capacity and facilities for development and testing of propulsion systems was given. These facilities are excellent for development of propulsion systems but were not designed for laboratory-type basic research in propulsion. The facilities consist of:

(1) Static test tower East side 2,000,000-lb capacity.

(2) Static test tower West side 500,000-lb capability with modification could handle 800k for S-II tests.

(3) Component test area, vacuum box for gear test, and back-to-back gear tester for turbopumps.

(4) Powerplant single stand 250k.

(5) Powerplant stand 100k.

(6) Test cell C for O_2/H_2 engines up to 35k.

[redacted]

(7) Component test cells, hot and cold environment. Six smaller cells for orifice calibrations of gas generators.

(8) Cold calibration test stand, 2 test sides. Four more component test cells for turbopump or model-size rocket test such as Saturn deflector and tail heating test at 4,000 lbs.

There are about 400 engineers at MSFC involved in the design, test, and development of propulsion systems in addition to those involved in the fabrication and inspection of systems.

Vacuum start facilities for liquid and solid rocket altitude start tests. Two facilities, one of 24,000 cubic feet, and one of 15,000 cubic feet. Vacuums to 1.5mm of mercury.

Dynamic test facility for full-scale Saturn C-2 to be ready by 1963. The stand can test dynamics of the vehicle launcher combination with full simulated loading; can suspend the Saturn fully loaded to determine overall dynamic response and vibration modes; can be used for propellant loading tests and to train launch crews. Facility is designed for a 3,000,000-pound-thrust vehicle.

It was noted that MSFC designs, develops, and fabricates systems and equipment. Static test facilities available are: (1) 40k and 50k blast facilities, and (2) hazard test facilities, (3) vacuum facilities, (4) 300,000-pound vertical test facility, (5) dynamic shake facilities, (6) a 2,000,000-pound test stand is being fabricated, and many others.

d. <u>Lewis Research Center (LeRC)</u>.-

<u>Evaluation of Pulse-Type Attitude Control Motor for Apollo</u>.- Tests of 1-pound and 25-pound Marquardt pulse-type, bipropellant rockets will be conducted at LeRC. It has been determined that a small vacuum tank already available at LeRC will be satisfactory for total motor operating time up to 20 to 30 milliseconds. Engine transients and specific impulse performance will be investigated.

<u>Research Applicable to Apollo Propulsion Systems</u>.- The following research projects are in progress at LeRC.

(1) Pressurization Gas Requirements for Liquid Hydrogen Propellant Tank.

(2) Pressurization Systems for Hydrogen-Fluorine.

(3) Experimental Study of Propellant Pressurization and Expulsion Systems.

(4) Experimental Evaluation At Zero Gravity of Expulsion

ing Techniques for Cryogenic Propellants.

(5) Experimental Study of the Effect of Propellant Sloshing on Attitude Stabilization System Requirements.

(6) Investigation of the Effect of Radiation on Rocket Propellants.

(7) Analysis of Thermal Protection of Propellants for Space Application.

(8) Radiation Heat Balance for Uncooled Nozzles.

(9) Thrust Chambers - H_2-H_2.

(10) Small Storable Liquid Propellant Rocket Motors.

(11) Experimental Evaluation of Bipropellant Attitude Stabilization System for Simulated Lunar-Landing System.

(12) Chemical Ignition and Controlled Stop and Restart of Solid Propellant Motors.

(13) Generalized Zero Gravity Program.

(14) Zero Gravity Heat Transfer Experiment.

(15) Complete Systems Integration - H_2-F_2.

(16) Modulating Thrust for Retro Propulsion - H_2-F_2.

(17) Experimental Study of Thrust-Vector Location and Drift Associated with Retro Propulsion.

Rocket propellants that have potential use as propellants in space will be exposed to radiation and the chemical and physical effects will be observed, among which will be possible physical decomposition and exothermic decomposition.

The "eight ball" project not listed is an experimental study of the effect of propellant sloshing on attitude stabilization system requirements.

LeRC has a sea-level facility for hydrogen fluorine experiments. An altitude facility is under construction and is scheduled to be complete in late 1961, which will handle a vehicle approximately 30 feet in length and 14 feet in diameter.

Most of LeRC's new large propulsion facilities are being installed at Plum Brook, and pump facilities, in addition to nuclear facilities, are located there.

In a discussion of fluorine development status, it was felt that fluorine systems had not been developed enough to put in a manned system at this time. Fluorine, if sealed clean, can be stored. Fluorine builds up heat if decontaminated and will burn through metals. LeRC has had problems with burn-through of valves. Bell Aircraft Corp. has been able to pump fluorine.

e. Jet Propulsion Laboratory (JPL).-

Current interests of JPL in propulsion are centered around the development of units for use in unmanned spacecraft. Nevertheless, since most of the propulsion problems such as reliability, space storage, zero g starting in the space environment, etc., are common to both manned and unmanned systems, some of JPL's activities in this area may be of interest in the Apollo program.

The three projects of current interest involving the use of propulsion are Ranger, Mariner, and Surveyor. The Ranger spacecraft is to be injected from a parking orbit into lunar range trajectories by the Atlas-Agena System. The first two units are not intended to actually encounter the moon and hence contain no propulsion. Rangers 3, 4, and 5, which are scheduled for launch in 1962, are designed for a semisoft landing of an instrumented package on the moon. Ford-Aeronutronics will provide the solid retropropulsion unit used to abstract most of the moon-relative energy from the package. JPL will develop and supply the 50-pound-thrust monopropellant hydrazine midcourse correction propulsion unit. The propulsion system is designed to deliver a specific impulse of 230-pound second/pound at 190 psi chamber pressure and 44 to one expansion ratio. It weighs on the order of 30 pounds loaded, contains 13.7 pounds of propellant and will deliver a maximum velocity increment of approximately 120 feet per second to the nominally 800-pound Ranger spacecraft. Some of the design features are displayed in photographs of a full-scale mock-up and in the system schematic. Regulated helium is used to pump anhydrous hydrazine from a butyl rubber bladder into the decomposition chamber. A bipropellant start is attained by injecting a small slug of nitrogen tetroxide into the chamber simultaneously with the opening of the main propellant valve. The chamber nozzle is radiatively-cooled Haynes Alloy No. 25. Single-actuation explosive valves are used throughout. Attitude control is provided by four jet vanes. The onboard accelerometer supplies the shutoff signal which simultaneously closes the main propellant valve and locks off the helium supply tank. The development of the Ranger unit is nearing completion with the first simulated space environment static firing of a flight unit scheduled for next month.

The Mariner A is a Venus fly-by experiment to be fired from a parking orbit by the Atlas-Centaur system. The spacecraft midcourse correction unit, which is also to be designed and developed by JPL, will use the same basic rocket motor and propellant pumping system as the Ranger. The Mariner A propulsion unit, in addition to providing for the midcourse correction maneuver approximately 16 hours out from earth, is designed to allow for a long-term storage and restart experiment which is to take place 4 days after the Venus encounter, i.e., approximately 4 months after the first firing. Difference from the Ranger midcourse propulsion system includes provisions for a larger propellant mass (32 pounds) to supply a total velocity increment of approximately 180 feet per second to the proposed 1,100-pound spacecraft; a change to nitrogen as the pressurizing gas; a rechargeable nitrogen tetroxide starting cartridge; and a change to a solenoid valve to provide for repeated operation, with multiple explosive valves being retained to lock off the nitrogen supply. The first firing of Mariner A is scheduled for 1962. Several firings of a heavy-weight version of the Mariner A motor and start system have been completed. A test of a flight unit will be conducted the latter part of this year in which the system will be fired in a vacuum chamber, shut off, stored in the desert for 4 months, returned to the vacuum chamber and restarted.

The Mariner B program for a Mars experiment which is now in study phase will most likely employ another restartable monopropellant propulsion unit of somewhat larger thrust, approximately 200 pounds.

The Surveyor project is directed toward soft landing a 200-pound scientific instrumented package on the moon using a 2,500-pound spacecraft launched by the Atlas-Centaur. Six-month studies of the complete spacecraft including provisions for midcourse-correction propulsion and for the retropropulsion have recently been completed by each of four outside contractors.

The current Advanced Development program efforts in solid propulsion are directed toward investigation of secondary injection of gases into the nozzle in order to achieve thrust vector control. A new six-component thrust stand and vacuum diffuser will be used to facilitate the experimental work. This follows an experimental flight program in which thrust reduction and thrust termination were successfully demonstrated by separation of the nozzle expansion cone and the head end of a solid propellant chamber, respectively.

In liquid propulsion, an Advanced Development program is underway wherein it is intended to ultimately provide a demonstration of a highly reliable and versatile bipropellant spacecraft propulsion system based on the use of nitrogen tetroxide and hydrazine or hydrazine derivatives. The present concepts of such a device are illustrated in a schematic of the proposed system. Both propellants are to be pumped from separate bladders located in a common tank and pumped by means of a single monopropellant gas generator. The hydrazine generant is

[redacted]

pumped from a prepressurized, bladdered tank. The use of a radiatively cooled chamber, possibly of pyrographite material, is to be investigated along with an injector design permitting simple on-off operation and throttleability. The test phases of the program are in the rudimentary stages at present. Water-pumping tests with a thin Teflon bladder have demonstrated the feasibility of using this material which is compatible with nitrogen tetroxide. Other materials will also be investigated for use as bladders. Small-scale pyrographite chambers have been procured but as yet are untested.

A comparative study on the use of various propellant combinations for the retropropulsion in a Mars-orbiter spacecraft launched by a Saturn booster (approximately 5,000-pound-spacecraft mass) is now nearing completion and may be of interest here. The systems compared are oxygen-hydrogen, fluorine-hydrazine or derivative, nitrogen tetroxide-hydrazine and an advanced solid propellant. In conjunction with this study, several papers on the space storage and venting of liquid propellants have been generated which may be of value to the problems at hand.

f. NASA Headquarters.- The following industrial contracts were let by NASA Headquarters:

(1) A study for hardware requirements for space missions and lunar landings. The study covers such aspects as requirements for throttleability, thrust to weight ratios; where do you sit on the moon, hot or cold side, and the effects of this on propellants.

(2) Two contracts will be let for experimental studies of variable thrust engines.

(3) A hardware contract on pressurization systems.

(4) A study contract has been let to A. D. Little Company on the storing of propellants in space. This will consist of a literature search on radiation and micrometeorite effects. LeRC is monitoring this contract.

It was stated that NASA Headquarters had compiled a document of the work being done at all of the laboratories and had distributed this to each of the Centers.

It was stated that the F-1 booster was progressing as well as could be expected without any specific requirement for the booster.

The following NASA Headquarters work is being done in solids:

[redacted]

[redacted]

(1) A study is being made to check the modulations in solid propellants by the use of a whistle technique.

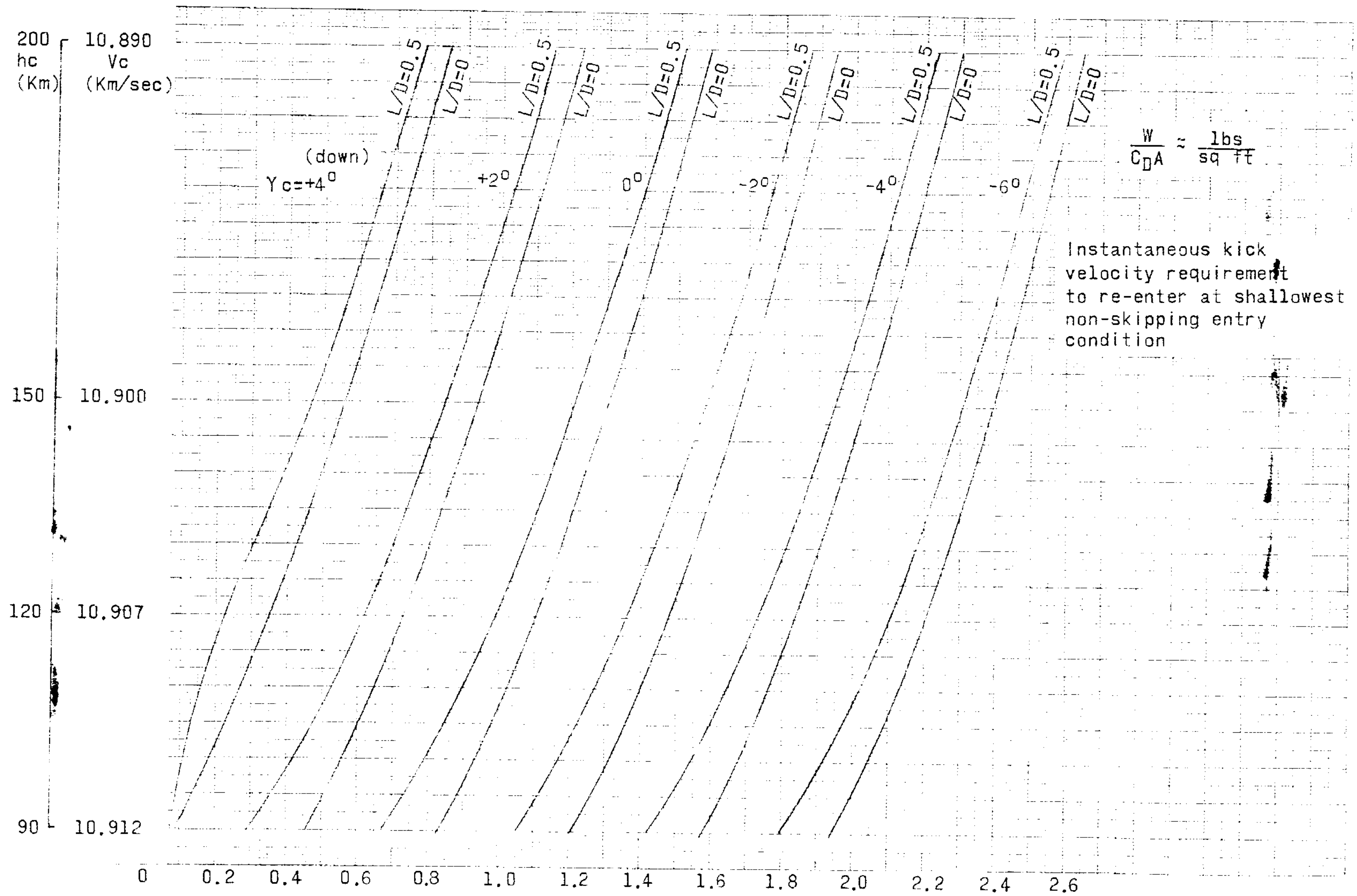
(2) Grand Central Rocket Co. has a contract to develop a reverse flow fiber-glass rocket motor called "wolf" (or flow spelled backwards).

(3) Large solid propellant booster studies are being made. One study consists of a modular construction with a taper. The group felt that a taper should be discouraged from the standpoint of complications involved in clustering tapered units.

(4) Velocity control and vectoring systems will be applied to the Scout vehicle. These will include gas generators and auxiliary solids.

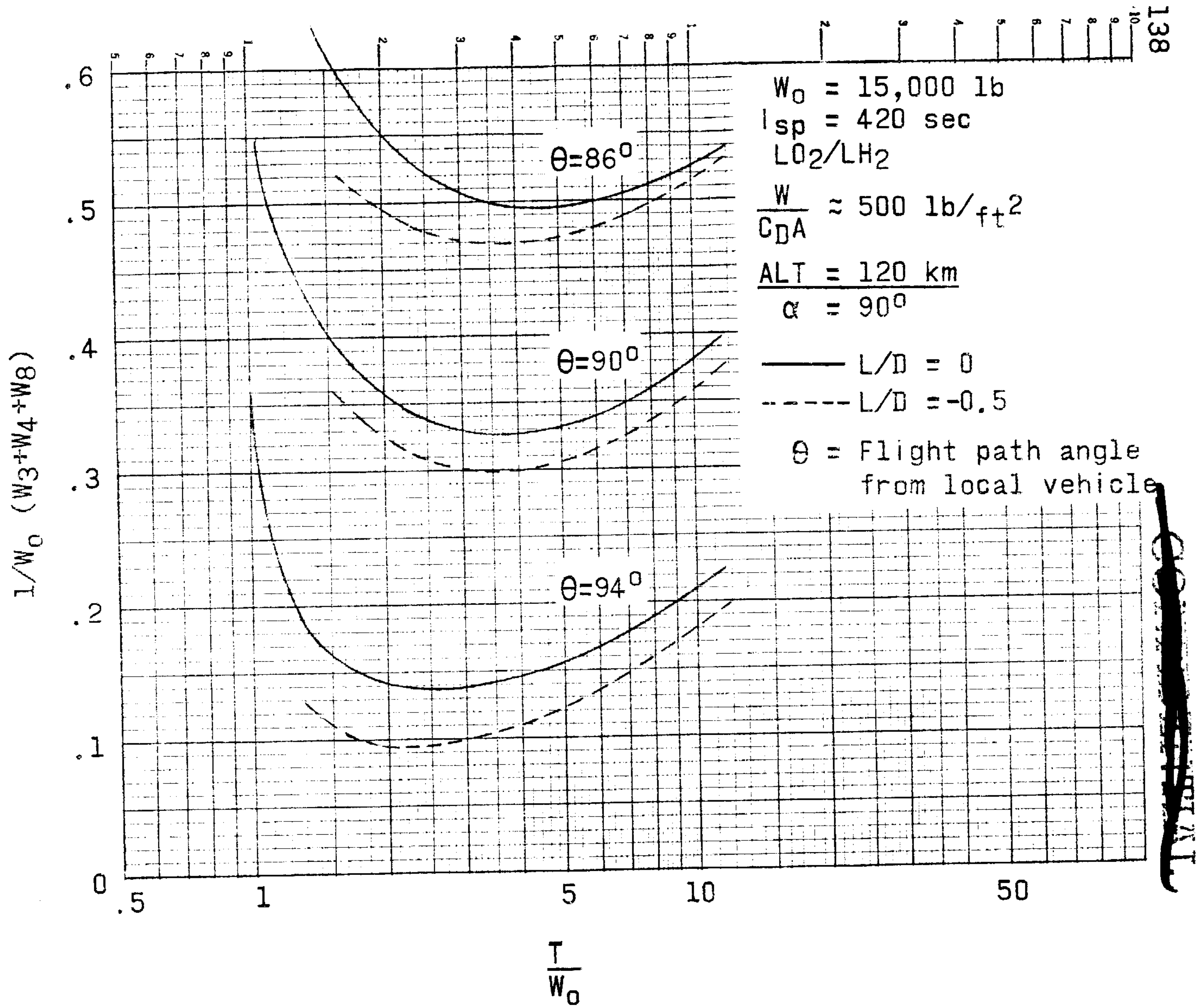
[redacted]

[redacted]



Section VIII. Figure 1.- Minimum kick required to reenter at compatible condition ΔV_K min (Km/sec) shallowest nonskipping.

[redacted]



Section VIII, Figure 2.- Relative weight of abort propulsion system required for shallowest nonskipping entry from lunar mission velocity.

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SECTION IX

EXCERPTS FROM

PROJECT APOLLO MINUTES OF MEETING

OF

TECHNICAL LIAISON GROUP MEETING NO. 1, STRUCTURES AND MATERIALS

January 12, 1961

Ames Research Center

Moffett Field, California

[redacted]

Reports from group members on Apollo-related activities at the various centers are as follows:

a. Ames Research Center (ARC).-

(1) Radiative properties of materials suitable for temperature control of vehicle -

(a) The effect of sputtering on the radiative characteristics of metal surfaces is being studied. In these studies, bombardment with hydrogen ions is used for sputtering copper, aluminum, and nickel targets. Both reflectance and emittance of the sputtered specimens are being measured.

(b) A flight experiment on evaluation in space of the time-dependent and environment-dependent changes in radiative properties of typical surface coatings is also underway. The flight package has been assembled and sent to Ball Brothers Research Laboratory for final checkout. The package is scheduled for flight on the S-16 solar spectroscopy satellite which is to be launched in the second quarter of 1961.

(c) In near-moon orbits, the vehicle will be influenced by the surface temperature of the moon. The vehicle temperature-control system must be adaptive to lunar surface temperatures from about -250° F to +220° F.

(2) Particle bombardment of vehicle surfaces -

(a) Ion bombardment - In this program, various metals are being bombarded with the various ionized gases to determine sputtering yields. The metals are Al W, Mo Fe, Ni, and Cu. The gases are N^+, N_2^+, H^+, H_2^+, and A^+. The sputtering yield is defined as the ratio of atoms of metal sputtered per impining ion.

(b) Meteoroid impact - One study is on the basic physics of hypervelocity impact in semi-infinite targets.

(c) One study is on the impact of particles into stony materials. In this program, attempts are being made to explain the meteoric craters on earth and on the moon.

(d) Micrometeoroid impact on multiwall shells has also been studied. In these studies, targets were impacted with glass balls to simulate stony meteoroids. The targets were single wall, double wall, and double wall filled with fiber glass. It was noted that the double wall with fiber glass was almost five times as efficient as the single wall of same total metal thickness. For this case, the weight penalty due to the fiber glass was about 30 percent.

(3) Ablation cooling -

(a) This is a new program that utilizes the combination of an arc-jet wind tunnel and an arc-image furnace for testing. With this equipment, studies will be made of the effects of combined radiative and convective heat fluxes on the ablative properties of plastics.

(b) In this program, tests were made to determine the effective heat of ablation of plastics at heating rates from 30 to 80 Btu/ft^2-second. The test specimens were made of two materials: Teflon and high-density polyethylene. It was found that for Teflon (a subliming material) the effective heat of ablation was about the same as that previously determined by others at much higher heating rates. For polyethylene (a melting and vaporizing ablator), the effective heat of ablation was only about 60 percent of that at higher heat rates. It is felt that this reduction is due to the melting (with consequent runoff of the liquid) of polyethylene.

(c) This program has no results at present. The tests include subjecting specimens to both high vacuum pressures and solar radiation.

(4) New equipment for ablation studies - The parabolic entry simulator, presently nearing completion, is designed to extend the studies on ablation cooling previously made in the atmosphere entry simulator to velocities approaching escape speed.

(5) Film cooling -

(a) An analysis of the effectiveness of helium film cooling in protecting a surface from convective heating has been made and a report on the results is almost completed.

(b) An experimental study of helium film cooling has just been completed. These tests were run in the 1-foot hypervelocity shock tunnel at a Mach number of 10, an enthalpy of 4,000 Btu/lb, and at a stagnation pressure of 4,000 psi. The results of these tests are presently being evaluated.

b. Jet Propulsion Laboratory (JPL).- JPL is doing no work which is directly related to Apollo. They are interested in flange design for high-vacuum seals and lubrication problems for low-temperature work. They are also conducting studies on a Mars-probe vehicle, visualizing a close approach in the early missions followed by a soft landing (50 fps landing velocity).

c. Langley Research Center (LRC).- LRC efforts fall into three broad areas. These are Thermal Protection Systems, Reentry Flight Tests, and Structural Dynamics.

(1) Thermal protection systems -

(a) Analysis of system efficiency - An analytical program has been underway to define the thermal efficiency of protection systems which can perform satisfactorily in the Apollo reentry environment. This work has shown that lightest weight is achieved by shielding systems which radiate a major fraction of the heat load at the shield surface and which incorporate an efficient mechanism for dissipating the fraction that is conducted into the interior. The mechanisms being studied are an endothermic phase change or decomposition which occurs at a low temperature in the shield interior and a water-boiling system at the shield back surface.

The analysis has permitted estimation of protection system weights for the range of heating conditions likely to be encountered in Apollo reentry trajectories. For typical vehicles, the indicated weight averages 4 to 5 pounds per square foot of vehicle surface area.

A machine computation routine for detailed analysis of ablation mechanism in deep-charring plastic ablators is being checked against available experimental data from hot gas jet tests. This work should permit an accurate prediction of the behavior of the charring ablators under long-time heating inputs.

(b) Ablation tests - An experimental program in air-heated airstream facilities has been underway to evaluate the relative performance of candidate heat-shield materials. Shield materials of equal weight (3 lb/ft^2) are exposed to the same cold wall heating rate (100 Btu/ft^2-second) and stream conditions ($T_s = 9,500^\circ$ R). The specimens are evaluated on the basis of the rate of temperature rise of a sensor mounted on the back face of the specimen.

In these tests, the deep-charring plastic ablation materials were markedly better than the noncharring plastic ablators. The length of time before the back surface reached 200° F for a deep-charring ablator (for example, phenolic-nylon) was 275 seconds as compared to 55 seconds and 80 seconds for noncharring ablators, Teflon and nylon respectively, and 80 seconds for glass-reinforced charring ablators. The radiation by the high-temperature char in conjunction with the low temperature of the ablating surface at the base of the char are largely responsible for this favorable behavior. Thermal efficiencies (total heat load divided by original shield weight) greater than 8,000 Btu/lb have been obtained with such materials at a stream enthalpy of 5,000 Btu/lb, whereas Teflon and nylon dissipate only about 3,000 Btu/lb at this enthalpy level. This program also includes tests of porous ceramics filled with various resins for comparison with the deep-charring materials.

(c) Erosion shields - Afterbodies of the Apollo reentry module may make use of nonablating radiating structures. The outer layer of this structure is known as an erosion shield and serves to confine an insulation layer.

Tests have been underway on several designs of refractory metal erosion shields. These tests have shown that rather simple shield designs can be attached to the primary structure in a manner that permits thermal expansion while avoiding aeroelastic difficulties. Depending upon the contours of the structure to be protected, the weight of such shields in molybdenum alloy should range from 0.8 to 1.2 pounds per square foot.

Oxidation tests on fabricated shields indicate that heating rates up to 50 Btu/ft^2-second can be tolerated for time periods in excess of the requirements of the Apollo reentry time. Tests are being conducted in arc-heated airstreams for comparison with tests in a furnace environment and also to provide a basis for comparison with future flight environment tests.

Mechanical and thermal property tests on several porous ceramic materials are also underway to determine their utility in an erosion shield application.

(d) Low-level cooling systems - Maintenance of a low interior wall temperature (< 150° F) is desirable in the reentry module. Various forms of water-cooling systems integrated with load-carrying structure are being tested for thermal performance as well as for effects on structural integrity. Tests have been completed on Bell Aircraft Corporation tubed sheet panels. A combined circulation-wick system of NASA design is under current investigation. The latter system is designed for a heat load up to 5 Btu/ft^2-second with localized rates up to 20 Btu/ft^2-second. Such rates are the highest anticipated heat leaks through external heat shields.

A supporting investigation is the calculation and measurement of heat-transfer properties of multiwall structures such as honeycomb and corrugated core sandwiches. A thermal conduction apparatus that can handle panels up to 2 feet square is near completion. This apparatus can supply heat flux rates up to about 20 Btu/ft^2-second in combination with pressure altitudes up to 180,000 feet.

(2) Reentry flight tests -

(a) Advanced materials and structural payloads - Materials payloads are under design for reentry tests at 30,000 fps. These payloads are scheduled for launch by Scout boosters starting in September 1961. A heating time of approximately 2 minutes is to be achieved on a shallow ballistic entry with data recovery by delayed-type playback prior to impact. The test area of the reentry shape is a blunt face approximately 11 inches in diameter. Data to be obtained are thermocouple readings throughout the cross-section of a charring ablator. A determination of char integrity is the primary goal.

In order to achieve longer test times in flight, an engineering study is being made of a lifting reentry body of a shape and size compatible with the Scout booster. The present configuration is a 5^{o} half-angle-cone cylinder about 10 feet long which achieves a heating time greater than 15 minutes from a release at 200,000 feet at 20,000 fps. This body is intended to be recoverable and is designed to exercise thermal protection systems in the Apollo flight corridor for realistic time periods. The structure incorporates a low-level cooling system and can accommodate either plastic ablation shields or metallic erosion shields.

(b) Heat transfer - Two heat-transfer payloads similar in body shape to the materials payloads are being designed for tests at 30,000 fps. One payload is to measure total heat transfer to a blunt face, whereas the other is to measure the radiative component. Particular interest enters on measurements made in the nonequilibrium flow region. The data from both of these payloads is to be obtained by delayed tape playback after emergence from the transmission blackout region.

Preliminary thinking is also going on concerning tests at 35 to 40,000 fps utilizing Minuteman boosters as well as available Jupiters. A small study contract has been let to the Chrysler Corporation to determine flight-test conditions obtainable with Jupiter.

(3) Structural dynamics -

(a) Booster structure - A $\frac{1}{5}$-scale dynamic replica of the Saturn SA-1 vehicle is under construction. The replica duplicates all important joints and fittings. The tanks will be capable of holding liquids, including liquid nitrogen in the lox tanks in order to reproduce temperature effects.

This model is to be used for vibration and stiffness studies of the Saturn booster inasmuch as confidence in analytical calculations for this complicated structure are in doubt. It is expected that stiffness and vibration testing will begin in mid-1961. Experience is currently being obtained in stiffness and vibration testing of clustered tank configurations using a model which is approximately a $\frac{1}{9}$-scale Saturn.

(b) Panel flutter - Accumulation of test data for establishment of flutter boundaries for various types of external skin panels is continuing. Recent work is directed at the effect of aerodynamic heating on flutter and the influence of pressure differential across the panel.

Some of the configurations tested are representative of erosion shield designs. Current plans are for tests of panels with curvature and which incorporate various degrees of flexibility in mounting to the base structure. A limited number of tests have shown that panel curvature is very helpful in preventing flutter.

(c) Particle impact - Data has been accumulated on penetration characteristics at speeds up to 16,000 fps. This work, as well as tests of the effectiveness of meteoroid bumpers, has been prepared for publication. Current tests in the gun facilities are being carried out at speeds of 26,000 to 27,000 fps using particles fired from opposing guns. A second light-gas gun is under construction to increase test speeds to 30,000 fps.

The S-55 micrometeorite satellite prototype has passed all developmental and environmental testing. The flight payload is being assembled for a planned launch in May 1961.

Calculations indicate that all of the different detectors should have been activated during the design orbit lifetime of 1 year.

A flight-test proposal, awaiting NASA Headquarters approval, is the launch of a paraglider with a 400-square-foot exposed area to obtain direct measurements on the frequency of input of small meteoroids and the resulting penetration depths. A probe shot of 7-minute duration in space is planned, and calculations indicate the occurrence of 3,000 to 4,000 measurable impacts. It is planned that the inflated glider will return to earth and be air snatched prior to impact. Examination of the glider should reveal information on the mechanism of cratering at high velocities and may lead to recovery of meteoroid material for laboratory analysis.

(d) Landing impact - Dynamic model tests have been carried out with several of the proposed reentry module shapes. The problem of dissipating horizontal velocity has been studied in particular. Whereas rocking motions occur during sliding impacts on earth surfaces, violent skips are encountered during water landings with shapes such as the lenticular. It would appear that letdowns with devices such as a paraglider will be restricted to landings on fairly well prepared surfaces. Test programs of large-scale paraglider systems are in the planning stage and may involve airdrops at Edwards Air Force Base.

Work continues on vertical impact attenuation devices such as collapsible structure, air bags, and rocket systems. The interaction of rocket systems with assumed conditions on the lunar surface are being studied in model tests.

d. Lewis Research Center (LeRC).- LeRC efforts fall into six broad areas. These are Meteoroid Damage Studies, Catastrophic Failure From High Speed Impact, Landing Impact Attenuation Analyses, Fracture Mechanics Studies, Superalloys Research, and Refractory Materials Research.

(1) Meteoroid damage studies - LeRC will have a meteoroid penetration experiment on the S-55 Scout satellite. LRC and MSFC will also have experiments on this same satellite. The LeRC experiment will measure penetrations that occur on 0.003-inch and 0.006-inch thick stainless-steel sheet with a total exposed area of a little less than 4 square feet. Penetration is recorded by breaking the continuity of gold-foil gages attached to the rear face of the stainless-steel sheets. There are a total of 60 sensors on the experiment.

A comprehensive study has been made at LeRC to catalog and interpret information that is presently available on meteoroids and damage that can be expected on space vehicles for near-earth missions. Based on this information, studies are being made to determine the type of experiments that should follow the S-55 satellite to obtain the most reliable information within the limitations imposed on satellite experiments.

(2) Catastrophic failure from high-speed impact - For some space vehicle structures, the danger from meteoroid impact may be far greater than that due to penetration of the structure. For materials that are already under a relatively high stress state, the impact of a meteoroid can initiate a complete structural failure that is in some ways related to the brittle fracture of pressure vessels. It is conceivable that this type of failure can be initiated without complete penetration of the structural element. The major concern on this type of failure is for pressure vessels such as the crew compartment, propellant tanks, or any portion of a pressure-stabilized structure.

As an extension of the fracture mechanics studies at LeRC in which failures of materials are studied under static load, a preliminary study has been started to obtain a better understanding of the failure mechanism resulting from high-speed impact on materials already under a static-stress condition. Preliminary studies have shown that catastrophic failure can occur even for materials normally considered quite tough. An understanding of the mechanisms affecting failure is being sought.

(3) Landing impact attenuation analyses - An analytical procedure was developed in Technical Report R-75 for studying the deceleration characteristics of landings on gas-filled bags. It was shown that for normal parachute descents, the deceleration and onset rates at earth impact are acceptable for well-supported humans. Additional analyses are being made to compare the effectiveness of this type of deceleration device with retrorockets and various shock absorption devices.

(4) Fracture mechanics studies - Brittle fracture is a serious problem in minimum-weight pressure vessels. In general, as material strength is increased in an effort to reduce the weight of material required, the material becomes more notch-sensitive and subject to failure resulting from stress concentrations or small material defects. Experimental studies are being made on the notch sensitivity of a wide variety of materials for a range of temperature down to that of liquid hydrogen. In addition, burst tests are being conducted on subscale pressure vessels for the same range of temperatures, and efforts are being made to correlate tensile specimen

tests with the strength potential of the same materials in pressure vessels. This information should be of use in designing pressurized structures in Apollo.

(5) Superalloys research - Research is being conducted on superalloys that have useful strengths for temperatures up to 2,000° F. Experimental alloys developed at LeRC have demonstrated high-temperature strengths considerably superior to commercial alloys presently available. Alloys of this type could possibly find use in areas subject to aerodynamic heating during reentry.

(6) Refractory materials research - Research is being conducted on metals and ceramics having the highest known melting points. These include tungsten (6,170° F), hafnium carbide (7,200° F) and tantalum carbide (700°.F). With further development these materials may find use on hot reentry structures. The research on tungsten is aimed at improving its workability and toughness. Research on surface finishes and improved purity have shown that under the proper conditions, marked increases in ductibility are possible. Very little is presently known about the physical properties of the two highest melting ceramics known (HfC and TaC). Experimental determination of their physical properties at temperatures up to 5,000° F is underway.

e. Marshall Space Flight Center (MSFC).-

(1) Structures - Presently at MSFC no work directly related to the Apollo capsule is underway in the field of structural design. However, investigations have been made or are underway with respect to the adaptability of the Saturn vehicle for use in the Apollo program.

(a) Load computation - Within these activities a preliminary load computation has been completed for a Saturn C-2 vehicle configuration, carrying the Apollo payload and using an S-I stage for 650,000 lbs of propellants, an S-II stage and an S-IV stage. Total vehicle length including Apollo was assumed to be 2,430 inches. On-pad and flight-loading conditions have been studied. The results have been summarized in a confidential NASA-MSFC internal memorandum, SD No. 18, dated November 15, 1960. This is preliminary data, but gives information about expected loads during "on-pad time" and during powered Stage-I flight. Available upon request from MSFC technical library or from MSFC, Structures and Mechanics Division, Structures Branch.

(b) Pressure connections - Flange connections of pressure lines, manhole covers of pressure vessels and other similar pressure connections in guided missile systems as well as other space vehicle structures, being subjected to large changes in temperature (in particular, temperature drops from room temperature down to liquid-oxygen or liquid-hydrogen temperatures) have been and still are trouble spots with respect to likelihood of the development of leaks and the resulting fire and explosion hazards. All efforts expended so far to develop connections with absolute pressure tightness resulted in certain improvements, but not in a 100-percent solution of the problem. At present, our outside contract work is planned at MSFC for a thorough investigation of this problem from the analytical and from the design (flange design as well as seal design) aspect. MSFC considers this leakage problem as especially severe for space structures since they have to operate not only under adverse environmental conditions such as vacuum and temperature environment, but have to stay tight over extremely large flight times compared to our present IRBM's and ICBM's.

(2) Materials - The following materials research work, applicable to a certain extent to the Apollo program, is either already accomplished or still underway in the Materials Branch of the Structures and Mechanics Division of MSFC. The report numbers and dates of publication are given with the research subjects below. These reports may be ordered through the Technical Library of AOMC, or the Technical Library of MSFC.

(a) Diffusion of gases through materials - It is well known that many gases will diffuse slowly through engineering materials. The diffusion rate is dependent not only upon temperature and pressure differential but also on the nature of the gas and material. Other factors being constant, diffusion rate is inversely proportional to the square root of the molecular weight of the gas. Hydrogen is the lowest molecular weight gas and is being considered widely as a propellant for chemical and nuclear rocket propulsion. For these reasons, primary attention has been placed on determination of the rate of diffusion of hydrogen through materials. Although considerable research has been reported on the mechanism of diffusion, almost all of this work was devoted to studies on pure materials. Very little experimental information is available on engineering materials.

After initial attempts to determine diffusion coefficients by measurement of pressure losses in a sealed container were unsuccessful, a method based upon mass spectrometric determination of gases diffusing through a membrane of the test material into a vacuum was developed. Accuracy and precision of this method were verified by comparison of experimental and literature values for the diffusion coefficient of hydrogen through palladium. The diffusion coefficients of hydrogen through nickel and type 301 stainless steel were found to be 1.9×10^{-16} and 3.6×10^{-17} cc (STP) sec/cm^2/mm thickness/mm pressure, respectively. The diffusion coefficients of aluminum and type 302 stainless steel were below 2.9×10^{-17} cc (STP) sec/cm^2/mm thickness/mm pressure, i.e., the limit of sensitivity of this method.

These results indicate loss of hydrogen by diffusion through aluminum or type 302 stainless steel containers for space applications probably will be negligible in comparison to ordinary leakages which usually are present. Assuming the most drastic conditions which may be expected, it is indicated that almost 4,000,000 years would be required to lose 1 percent (wt) of liquid H_2 by this process. However, the time required to significantly reduce the effectiveness of vacuum jacket insulation schemes will be far shorter than that for 1-percent weight loss to occur. This factor is much more critical since once insulation becomes markedly reduced, the boiloff rate of liquid hydrogen will increase greatly, thus leading to vessel rupture or excessive loss by venting. Again, using the most drastic conditions which can be anticipated, it can be calculated that the start of deterioration of vacuum-type insulation effectiveness will be between approximately 100 days and 7×10^{14} years, and will be essentially complete between 27.5 and 7×10^{16} years. The first value stated for each of the above cases (start and completion) are based upon the maximum diffusion coefficient of stainless steel or aluminum as determined experimentally in this program and the latter values are based on dubious extrapolations from the literature values of the diffusion coefficient for aluminum.

More sensitive methods of determination of diffusion rates of hydrogen through engineering materials are essential in order to predict more accurately the lifetime of insulation schemes employing vacuum jackets.

(b) Compatibility of engineering materials with space environment - One of the most important considerations for the selection of a material to be used in the space environment is compatibility with ultrahigh vacuum. For this reason, a means was developed for determining the evaporation rate of materials under conditions of high vacuum and temperature.

Studies on pure compounds showed that the theoretical explanation of the mechanism of evaporation of materials at elevated temperatures can be applied to the evaluation of the compatibility of engineering materials with reduced pressures at ambient temperatures. In the latter case, however, the rate of evaporation will decrease to a constant value because heat is not supplied to the sample to maintain the temperature at its initial value.

Extrapolation of data available in the literature on the evaporation rates of pure metal at elevated temperatures indicates that the evaporation of some metals in space may be of more importance than changes in mechanical properties in limiting their useful temperature range over an extended period of time.

Five vacuum systems were developed and tested for evaluating the vacuum compatibility of materials. The combination of these systems provided a capability of producing environments at pressures of 10^{-3} to 10^{-10} mm Hg and temperatures up to 150° C. By incorporating an electronic balance into a vacuum system, the rate of weight loss, i.e., rate of evaporation could be determined continuously.

The compatibility of materials with vacuum was investigated in a stepwise manner. Initial tests were made at ambient temperature in a system capable of reaching 10^{-5} mm Hg. Those materials which lost no appreciable weight under these conditions were retested at 50° C and 100° C in the same system. The best materials screened on the basis of test in this system were tested and culled similarly in systems capable of 10^{-6}, 10^{-7}, 10^{-8}, and finally 10^{-10} mm Hg.

Because the great number and variety of engineering materials which may be employed in the space environment precluded testing of all within a feasible time or effort, only those materials which were under active consideration for such use, and were suspected of having appreciable vapor pressures, were selected for experimental testing.

Experimental results indicate that Teflon probably is the most satisfactory of organic-type materials with respect to evaporation in vacuum. Mylar, with or without an aluminum coating, is satisfactory at temperatures below 100° C. Organic insulations on electrical wiring for space applications should be checked for vacuum compatibility. It was found in several instances that the plasticizer volatilized, causing severe embrittlement of the insulation and depositing an oily film on cooler surfaces where it condensed. Most lubricants partially volatilized in vacuum. This effect, however, is not necessarily detrimental. Thus, vacuum compatibility tests on lubricants preferably should be done while in use under simulated service conditions.

(c) Effect of space environment on the extreme pressure qualities of lubricants - The wear characteristics and load-carrying capacities of a petroleum oil and a synthetic oil, with various extreme pressure compounds added, were studied under boundary lubrication conditions at reduced atmospheric pressure and in an inert atmosphere. Results of this work indicated that the lubricating properties of the oils tested were not changed when operating in an atmosphere of nitrogen. However, the lubricating qualities of the petroleum-based oils were drastically reduced when subjected to an absolute pressure of 20 microns of mercury. The lubricity and load-carrying capacity of the synthetic-based lubricants were only slightly reduced at the lower atmospheric pressure. This work has shown that the loss of lubricants by evaporation is not the only problem associated with the reduction of atmospheric pressure, but that deterioration of wear characteristics and load-carrying capacity of lubricants can occur at environmental pressure higher than where evaporation would become a serious problem.

(d) Effects of space environment on certain physical and chemical properties of materials - (Report No. MIP-M-MS-IP-60-1, being published at the present time.)

The effects of the space environment on materials must be studied and understood before highly reliable space flight can be realized. Since it is impractical, if not impossible, to create the total environment of space in the laboratory, it is necessary to study the effects of those components of the space environment which can be simulated in the laboratory and especially those which are expected to have the greatest detrimental effects on materials. The vacuum component of space is probably the most easily simulated, and is believed to be the major contributor to the degradation of the inorganic materials. Therefore, the study of the effect of vacuum on materials constituted the major portion of this program.

The program was divided into three parts, defined as follows:

PART I - The Effect of High-Vacuum Conditions on the Fatigue Properties of Metals

PART II - The Effect of Vacuum on Lubrication

PART III - An Investigation of Bonding Forces Through Evaporation

The parts were studied individually and separate progress reports were prepared on each part. However, the results of each part were correlated with those of the other. PART III was designed to support PARTS I and II.

<u>PART I</u> - An investigation to determine the influence of reduced pressures on the fatigue properties of several metals at room temperature indicated that, by the reduction of oxygen, the fatigue properties of some metals can be improved. Fatigue life under vacuum conditions may be ten times greater than at corresponding stress levels in air. Increased fatigue properties in reduced pressures depend upon the oxygen affinity of the particular metal. By the elimination of oxide film formation on crack walls or the reduction of oxygen diffusion at the crack front, the rate of crack propagation can be substantially reduced by vacuum conditions.

<u>PART II</u> - The parameters which contribute to successful lubrication in the space environment were investigated. Special equipment, designed specifically for this purpose, permitted the coefficient of friction of a running bearing to be monitored for the life of the bearing while exposed to a pressure of 10^{-6} mm of mercury. Electroplated silver films, in varying thicknesses, were used exclusively in the experimental program in order to evaluate an optimum thickness of the lubricant film. Other parameters studied included: (1) surface finish, (2) surface cleanliness, (3) adherence of lubricating film to bearing, and (4) tolerance between rotating bearing components. Initial testing considered pure sliding friction as experienced in sleeve bearings with later work devoted to combined sliding and rolling friction as experienced in ball bearings.

It should be noted that this study did not include any petroleum base or synthetic-based lubricants as studied under ARPA order 92-59.

PART III - This investigation was undertaken to measure the bonding force between oxide and metal by a method which avoids the utilization of mechanical testing devices. The method is dependent upon the variation of the evaporation rate, as a function of film thickness, of a thin metal film (few Angstroms in thickness) deposited on its oxide. The equipment used is described, and this equipment was designed so that the change in apparent vapor pressure with time could be determined. From this change in vapor pressure, the bonding energy between aluminum and aluminum oxide was determined.

The presentation of the results of these studies of the effects of the space environment on certain physical and chemical properties of materials does not imply that the problems have been solved; to the contrary, the effort has been successful primarily in defining the magnitude of the problem.